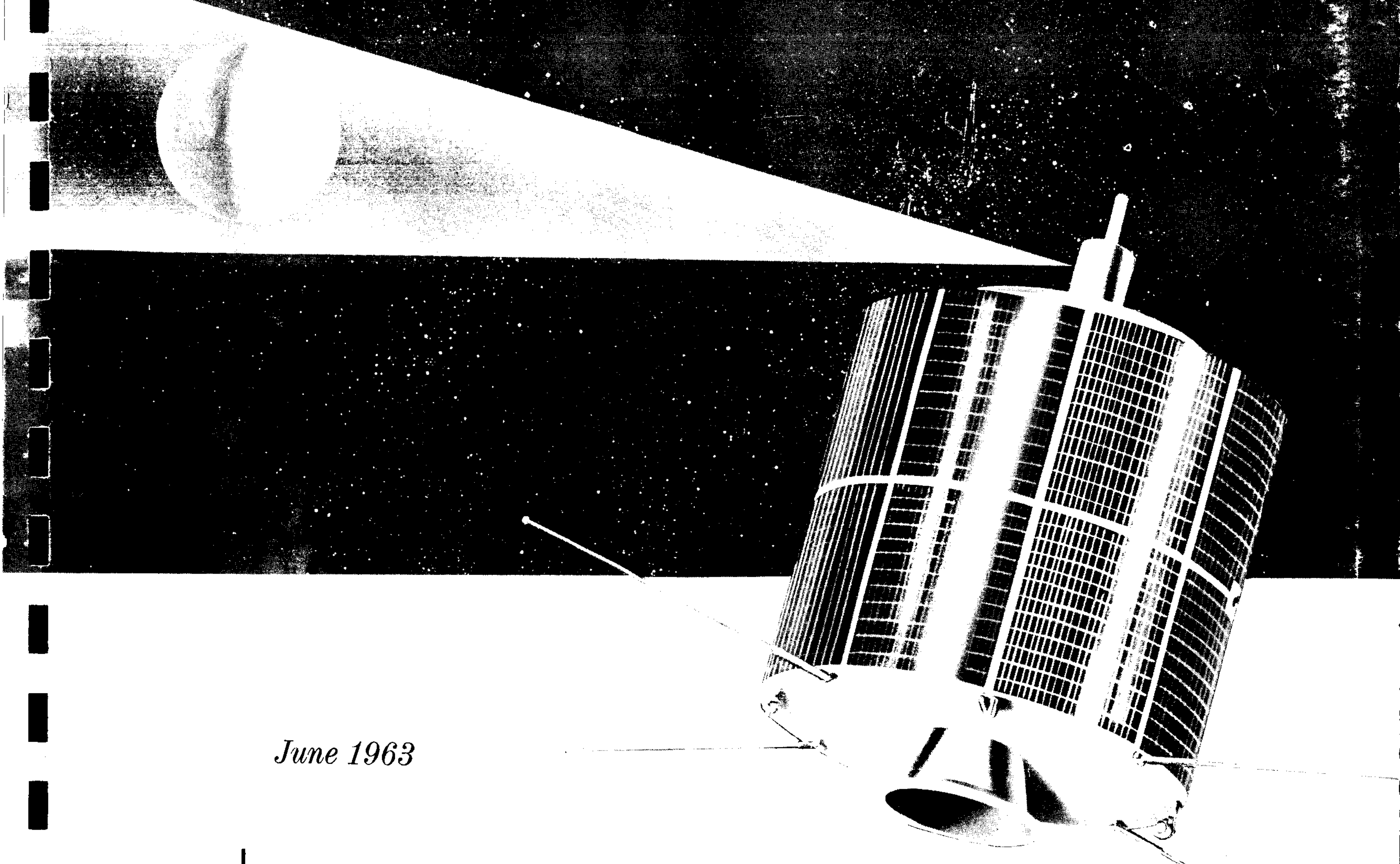
Advanced SYNCOM



June 1963

MONTHLY PROGRESS REPORT

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NASA Contract 5-2797
SSD 3401R

AEROSPACE GROUP
SPACE SYSTEMS DIVISION
HUGHES AIRCRAFT COMPANY
CULVER CITY, CALIFORNIA

HUGHES

HUGHES AIRCRAFT COMPANY

AEROSPACE GROUP
SPACE SYSTEMS DIVISION
EL SEGUNDO, CALIFORNIA

20 July 1963

SUBJECT: Advanced Syncom Monthly Progress Report for June 1963

TO: Mr. Robert J. Darcey
Program Manager, Syncom
Goddard Space Flight Center
Code 621
Greenbelt, Maryland

Attached are copies of the Advanced Syncom Monthly Progress Report for June 1963.

A simplified battery charging system has been devised which eliminates the need for a separate battery regulator. The proposed method uses a portion of the solar array to directly charge the battery. The solar array characteristics provide both voltage and current control. In addition, the portion of the array used for battery charging is available for other loads when the battery is not being charged.

The structural redesign detail work is continuing. The longitudinal stiffening members have been located, and the method of connecting the forward equipment structure to the thrust tube has been chosen. Target weights have been assigned to all subsystems to assist in maintaining weight control.

A communications test panel has been designed which incorporates a unique multiple-access receiver. In this receiver the input signal is divided down in frequency so that the modulation index is reduced to the same value as that utilized in the phase modulator in the spacecraft. This reduction in index permits the use of coherent detection in a simple phase detector.

HUGHES AIRCRAFT COMPANY

Paul E. Norsell

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HUGHES

CONTENTS

1. INTRODUCTION

The use of communication satellites has been recognized to answer the need for greatly expanded global communications capability. It has been a major effort of the United States Government and of industry to develop a satellite relay system at the earliest possible time.

Under NASA Goddard Space Flight Center Contract NAS-5-1560, Hughes Aircraft Company developed the Syncom I spacecraft to be orbited by NASA Delta launch vehicles and used in conjunction with Department of Defense Advent ground stations for the performance of inclined synchronous-orbit communication experiments during 1963.

The Syncom I spacecraft will demonstrate a simple spin-stabilized design capable of being placed in a synchronous orbit. At the same time, it will be demonstrated that a simple pulse-jet control system can provide the stationkeeping necessary to maintain a synchronous orbit.

Additional important mission objectives of the NASA communication satellite program include the demonstration of a "stationary" or equatorial, synchronous orbit, conduct of system orbital life tests, demonstration of new wide-band services on a transoceanic basis, and demonstration of a system accessible to all nations.

Under NASA Goddard Space Flight Center Contract NAS-5-2797, Hughes is conducting feasibility studies and advanced technological development for an advanced, stationary, active repeater communication satellite. A Summary Report covered the technical progress achieved during the original contract period and detailed the system configuration resulting from the system studies. A subsequent supplementary report covered further studies made under modification two to the above contract and the accompanying technical direction.

This is the second report under modification three to the above contract and covers activities during the month of June 1963.

2. COMMUNICATION SYSTEM DESIGN

A simplified block diagram of the present design of the multiple access transponder and of the multiple access test panel is shown in Figure 2-1. In this form the basic frequency relations and the similarity of units in the two equipments are readily apparent. In addition to serving as a test panel, the circuit in Figure 2-1 is also suitable for use in a communication terminal. Several advantages accrue from the use of this configuration. The spacecraft master oscillator becomes the system master oscillator, and no frequency synthesizer is required for the ground terminals. The use of the complementary frequency divider reduces the spectrum to the low modulation index signal corresponding to the phase modulator output in the spacecraft. This permits the use of coherent detection in a simple phase detector, avoiding the necessity of an elaborate linear frequency discriminator. If necessary, the communication signals can be recovered completely free of incidental phase modulation produced by the low-frequency phase noise spectrum of the spacecraft master oscillator. Since the ground oscillator follows these low-frequency phase disturbances, the 6-kmc chain output has the same phase noise as the spacecraft local oscillator, delayed by 1/8 second. This delay is exactly equal to the delay on the communication signals measured from the time the noise is impressed on them at the spacecraft mixer. A signal is therefore available to compensate exactly for phase instability in the spacecraft, without recourse to a retransmitted pilot signal. The lowest frequency part of the spectrum is automatically nulled. If noise frequencies above a few cycles per second become important, they can be measured separately and cancelled from the received spectrum.

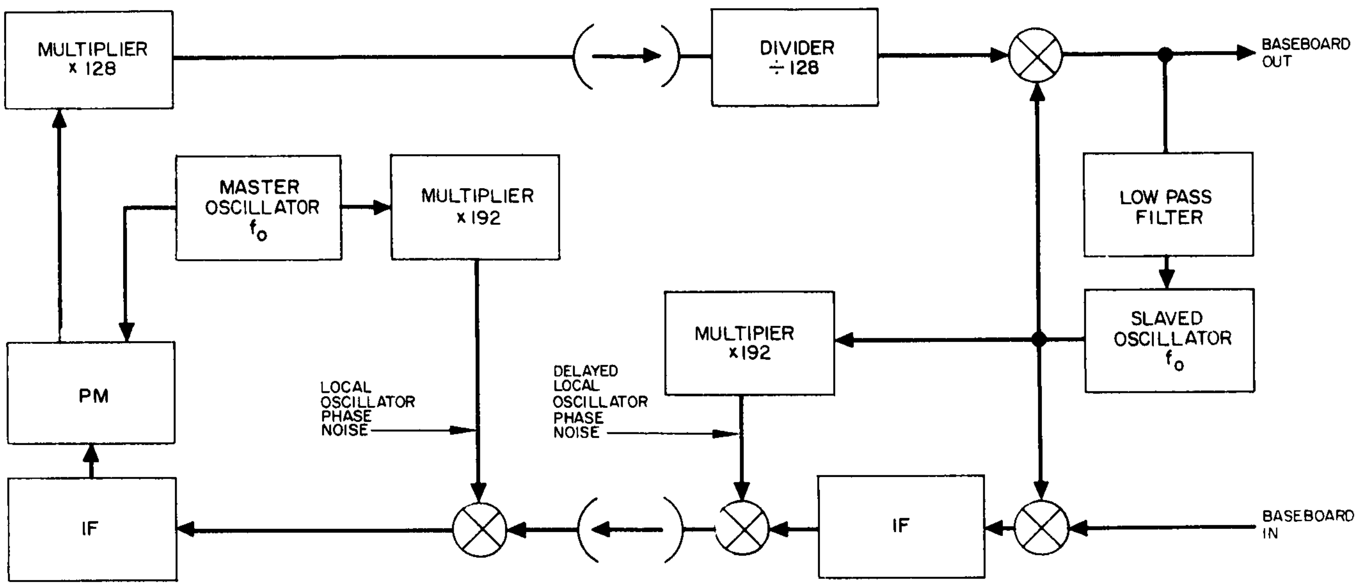

Figure 2-1. Simplified Block Diagram of Multiple-Access Transponder and Multiple-Access Test Panel

3. LAUNCH AND ORBIT ANALYSIS

TRANSLATION DUE TO INITIAL ORIENTATION IMPULSE

During an orientation maneuver, the axial jet produces simultaneously both a torque and a translatory impulse. Since any translation of the spacecraft represents a potential need for stationkeeping control action, it is desirable to estimate the magnitude of the translation impulse.

The greatest number of axial pulses fired in one sequence is expected during initial orientation (~1250 pulses, calculated by means of the orientation dynamics IBM computer model*). Thus, the calculation below assumes this number of pulses.

Starting with the basic assumptions that the spin axis traces a great circle instead of a rhumb line, and the large number of discrete jet pulses closely approaches a continuum, the N-component of the translation impulse is 90 degrees (Figure 3-1)

$$\int_{\theta_o} \Delta V_o \sin \theta d\theta$$

which for θ_o (the inclination of the spin axis prior to orientation) = 24.4 degrees and ΔV_o = 18/1.145 = 15.72 fps/radian is 15.72 x 0.911 = 14.3 fps. (ΔV = 18 fps is the equivalent ΔV required of the control system for the orientation maneuver, and 90 degrees - 24.4 degrees = 65.6 degrees = 1.145 radians is the precession angle.) It should be noted that this impulse is always in a northerly direction.

14.3 fps is ~12 percent of the estimated ΔV (112 fps, 3σ) required to correct for ascent guidance inclination errors, and hence will be taken into account in estimating total ΔV requirements for the mission. Similarly,

$$\Delta V_{east} = \int_{\theta_o}^{\pi/2} \Delta V_o \cos \theta \, d\theta = \Delta V_o (1 - \sin 24.4 \text{ degrees})$$

$$= (15.72)(0.587)$$

$$= 9.23 \text{ fps}$$

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*"Advanced Syncom Monthly Progress Report," Hughes Aircraft Company, May 1963.

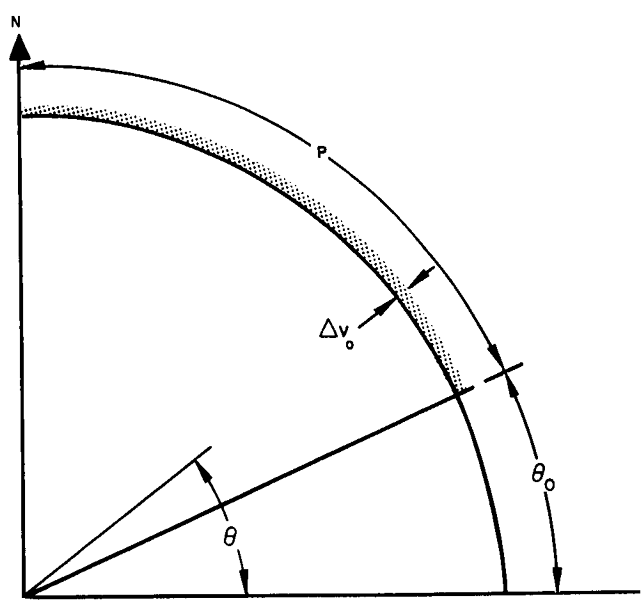

Figure 3-1. Translation Impulse Due to Initial Orientation Impulse

INDUCED NUTATION ANGLE DURING PULSED OPERATION OF AXIAL JET FOR INCLINATION REMOVAL

In the May Progress Report, pages 3-6 to 3-8, it was shown that the reduction of the axial jet duty cycle to one-third (due to possible axial jet temperature rise problems) resulted in a negligible effect on the efficiency of the inclination removal when compared with the continuous operation sequence.

This report presents an analytical model of the induced nutation angle history, $\theta(t)$, during axial jet pulsing for inclination removal. The effect of a simplified first-order nutation damper time constant, τ_d, is also included. Numerical substitution of typical design parameters indicates a maximum induced nutation angle, $|\theta|$ max, of about 1.7 degrees during a 54-minute inclination removal sequence of one-sixth duty cycle with 12 seconds "on" pulse time. This angle may be further reduced if the spin phase angle at which the jets are turned on and off is shifted a number of times during the sequence in a manner that will tend to cancel the nutation angle excited from previous pulses.

Mathematical Model

Considering the spacecraft as a spinning symmetrical rigid body with constant spin speed, ω_z, spin axis moment of inertia greater than the transverse moment of inertia, $I_z > I_x = I_y$, the Euler equations of motion (in body coordinates x, y, z) resulting from axial jet pulsing may be represented in complex form by $\left(\text{note: } \dot{\omega} = \frac{d}{dt}\omega\right)$

$$\dot{\omega} + \Omega_c\, \omega \approx n \tag{3-1}$$

$$\Omega_c \equiv \frac{a}{\tau_d} + j\,\Omega\,;\; j = \sqrt{-1} \tag{3-2}$$

$$\Omega \equiv \left(1 - \frac{I_z}{I_x}\right)\omega_z \equiv \omega_z - \omega_n \equiv \left(1 - \frac{I_z}{I_y}\right)\frac{2\pi}{\tau_s} \tag{3-3}$$

$$n = \begin{Bmatrix} n_o = \dfrac{N_o}{I_x};\; q(m+k)\tau_s \le t \le [q(m+k)+k]\,\tau_s \\[2ex] 0 \qquad ;\; [q(m+k)+k]\tau_s \le t \le (q+1)(m+k)\,\tau_s \end{Bmatrix};$$

$$q = 0, 1, 2, \ldots, \nu-1 \tag{3-4}$$

where

$\omega = \omega_x + j\omega_y$

$=$ component of body angular velocity normal to spin axis, radians per second

$a =$ dimensionless constant (= 1 in this example)

$\tau_d =$ nutation damper time constant, seconds
$\approx$ 300 seconds

$\tau_s =$ spin period seconds
$\approx$ 0.6 seconds $= 2\pi/\omega_z$

$\omega_n =$ nutation frequency (undamped), seconds

$\cong \frac{I_z}{I_k}\omega_z$

$n =$ normalized torque, ft-lb/slug-ft^2

$N_o =$ applied torque
$\approx$ 10 ft-lb

$I_x =$ transverse moment of inertia, slug-ft^2
$\approx$ 51.7 slug-ft^2

$m =$ number of "jet-off" spin periods per on-off cycle
$= 0, 1, 2, \ldots$

$k =$ number of "jet on" spin periods per on-off cycle

Duty cycle $= \frac{k}{m + k}$

$q =$ running index of on-off cycles

$= 0, 1, 2, \ldots, \nu-1$

$\nu =$ number of on-off cycles required to remove an initial inclination, Δi

$= \frac{t_a}{k\tau_s}$; t_a = 544 seconds for Δi = 0.65 degrees*

*"Advanced Syncom Monthly Progress Report," Hughes Aircraft Company, May 1963.

A solution to Equation 3-1 for ω is given by

$$\omega = \omega_o e^{-\Omega_c \tau} + \frac{n}{\Omega_c} \frac{\left(1 - e^{-\Omega_c k\tau_s}\right)\left(1 - e^{-\Omega_c \nu(m+k)\tau_s}\right)}{\left(1 - e^{-\Omega_c (m+k)\tau_s}\right)} \tag{3-5}$$

where

τ = total time of inclination removal process with a reduced duty cycle

$$= [\nu(m + k) - m] \tau_s = \left[\frac{t_a}{k\tau_s}(m + k) - m\right] \tau_s$$

ω_o = initial component of body angular velocity normal to the spin axis (assumed zero).

The subsequent nutation angle, θ, is therefore given by

$$\tan\theta \approx \theta = \frac{\omega}{\omega_n} \tag{3-6}$$

Equation 3-6 can be maximized by noting that in general

$$\nu(m + k) >> k$$

$$e^{-x} = 1 - x + \frac{x^2}{2} - \frac{x^3}{3!} + \ldots$$

and also, in this particular case,

$$\left|\frac{a}{\tau_d}\right| << |\Omega|, \quad (a \cong 1)$$

so that

$$|\Omega_c| \approx |\Omega|$$

Thus,

$$\max|\tan\theta| \approx \max|\theta| \approx \left|\frac{N_o \tau_s}{(2\pi)^2 I_z \left(1 - \frac{I_z}{I_x}\right)}\right| \left(\frac{2\tau_d - ak\tau_s}{a(m+k)}\right) \tag{3-7}$$

$$= \left|\frac{n}{\Omega_c}\right| \frac{1}{\omega_n} \left(\frac{2 - \frac{ak\tau_s}{\tau_d}}{a\,(m+k)\frac{\tau_s}{\tau_d}}\right)$$

If $k = 20$, $m = 100$ to yield a duty cycle of

$$\frac{k}{m+k} = \frac{1}{6}$$

and a total inclination removal time of

$$\tau = \left[\frac{544}{12}(120) - 100\right] 0.6 = 3204 \text{ seconds} = 53.4 \text{ minutes}$$

the maximum nutation angle becomes

$$\max|\theta| = \left(\frac{0.0527}{14.7}\right)\left[\frac{2 - (20)(0.002)}{(120)(0.002)}\right] = 0.0294 \text{ radian}$$

$$= 1.68 \text{ degrees}$$

This value of max $|\theta|$ as well as that of τ appear to be tolerable.

4. SPACECRAFT SYSTEMS DESIGN

SPACECRAFT SUBSYSTEMS PERFORMANCE REQUIREMENTS SPECIFICATION AND BLOCK DIAGRAM, REVISION B, 28 JUNE 1963

1.0* Introduction

1.1 Purpose: The purpose of this specification is to define requirements to which each subsystem of the Syncom II is to be designed and tested.

1.2 Scope: This specification defines what is required of each subsystem of the Syncom II.

2.0 Applicable Documents

2.1 The following documents form a part of this specification to the extent specified herein:

MIL-W-8160D	Dated 17 March 1961 Wiring, Guided Missile Installation of General Specification for
MIL-I-26600(USAF)	Dated 2 June 1958 Amendment 1, dated 17 June 1959 Interference Control Requirements Aeronautical Equipment
General Range Safety Plan Volume I, Missile Handling	Dated 1 April 1960, Errata Sheet Dated 4 May 1960, Revision 1 Dated July 1960, Revision 2
LMSC-A057612	Dated 30 September 1962 Syncom Booster Feasibility Study Final Design Report Lockheed Missile and Space Company

* The numbers in this section refer to the actual specification numbers.

Technical Memorandum 732	Dated October 1962 Environment of Syncom Mark II Paul M. Blair, Jr. and Herbert T. Toda
S2-0100	Dated 18 February 1962 Performance and Test Specification Advanced Syncom Spacecraft
	Dated 15 May 1963 Syncom II RF and Electrical Interface Specification
	Dated 15 May 1963 Syncom II Mechanical Interface Specification
NASA Document MSFC-PROC-158B	Dated 15 February 1963 Procedure for Soldering of Electrical Connectors

3.0 Requirements

3.1 Definition of Spacecraft Subsystems. The major and minor control items have been grouped together into functional groups as subsystems. These subsystems and the control items of which they are composed are listed below and shown diagrammatically in Figure 4-1.

1) Communication Subsystem
   475025, 475030, 475040

2) Antenna and Jet Control Subsystem
   475035, 475303, 475160

3) Telemetry and Command Subsystem
   475045, 475050, 475055

4) Power Supply Subsystem
   475060, 475251, 475252, 475253, Battery

5) Spacecraft Structure Subsystem
   475065, 475301, 475302, 475304, Separation Switch

6) Wire Harness Subsystem
   475300

7) Apogee Motor Subsystem

8) Reaction Control Subsystem

3.2 Communication Subsystem: The communication subsystem shall provide facilities to receive, convert frequency, amplify, and retransmit microwave signals. Each transponder shall be capable of operating in either a frequency-translation mode or a multiple-access mode. An unmodulated beacon signal shall be transmitted to provide a signal for ground antenna autotrack. Series and/or paralleled redundant units shall be used as necessary (consistent with weight and volume limitations) to satisfy reliability requirements. The communication subsystem shall be composed of:

1) Four Communication Transponders, 475025

2) One Communication Antenna, 475030

3) Four Communication Transmitters, 475040

3.2.1 Reliability: The communication subsystem shall have a probability of operation within the performance requirements of 0.906 for a 1-year requirement and 0.525 for a 3-year requirement.

3.2.2 Communication Transponder - 475025

3.2.2.1 Quantity: There shall be four communication transponders.

3.2.2.2 Modes: Each transponder shall be capable of operating in either the frequency-translation or the multiple-access mode.

3.2.2.3 Frequency Assignments: The frequency assignment for transponders shall be (in megacycles) as given in Table 4-1.

3.2.2.4 Common Requirements: Each mode of the transponder shall meet the following requirements.

3.2.2.4.1 Input and Output Impedance: The input and output impedance of each receiver of the transponder shall be approximately 50 ohms.

3.2.2.4.2 Noise Figure: The noise figure of each receiver shall be better than 9 db (referenced to the standard noise temperature of 290°K).

3.2.2.4.3 Power Out: Each receiver shall have a power out of 1 mw ± ____mw.*

3.2.2.4.4 Telemetry Outputs: Each receiver shall provide an output from the IF strip for transmission by telemetry transmitter.

3.2.2.4.5 Transponder Power: Each transponder shall require no more than 75 ma at 24 volts.

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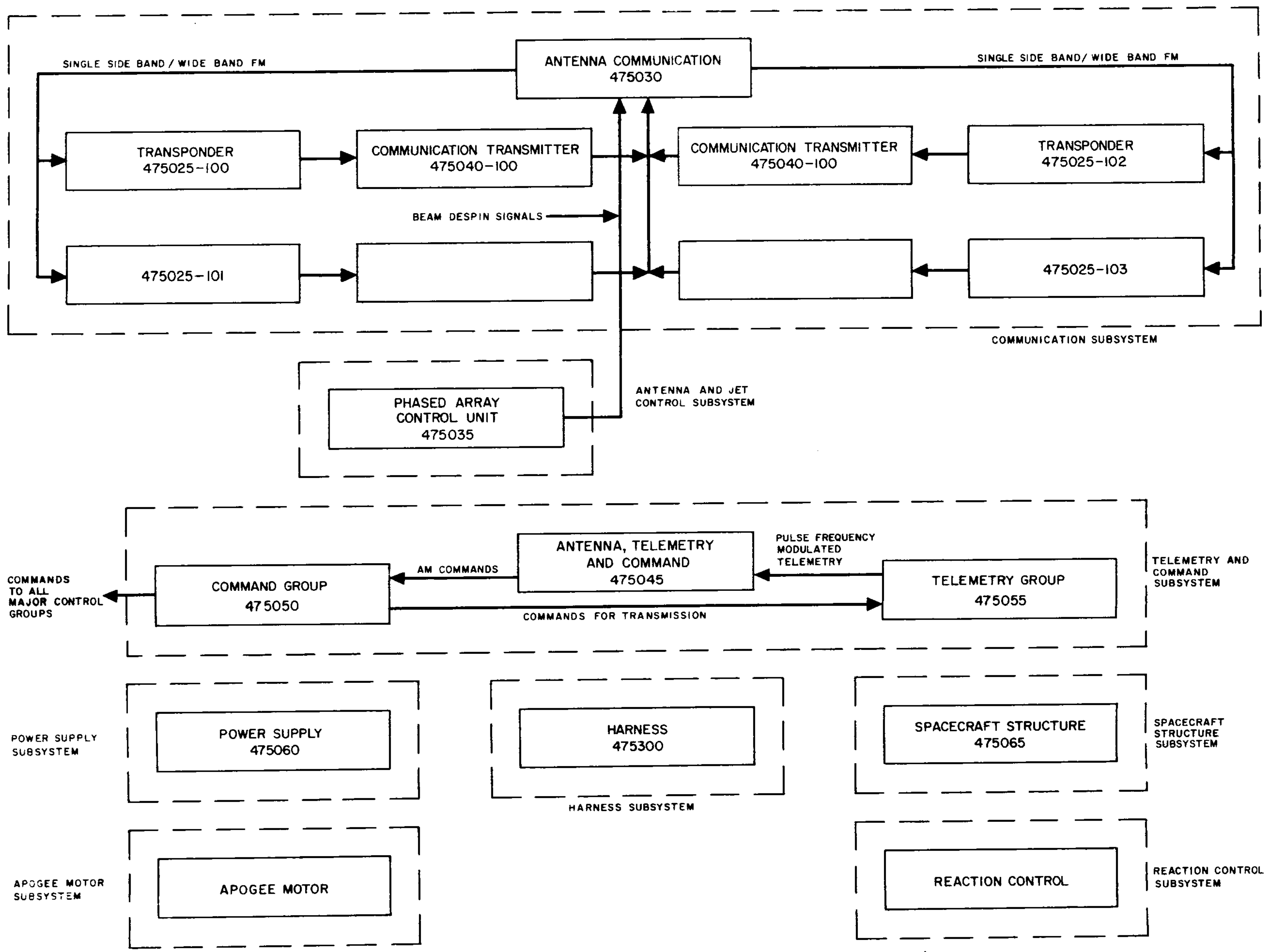
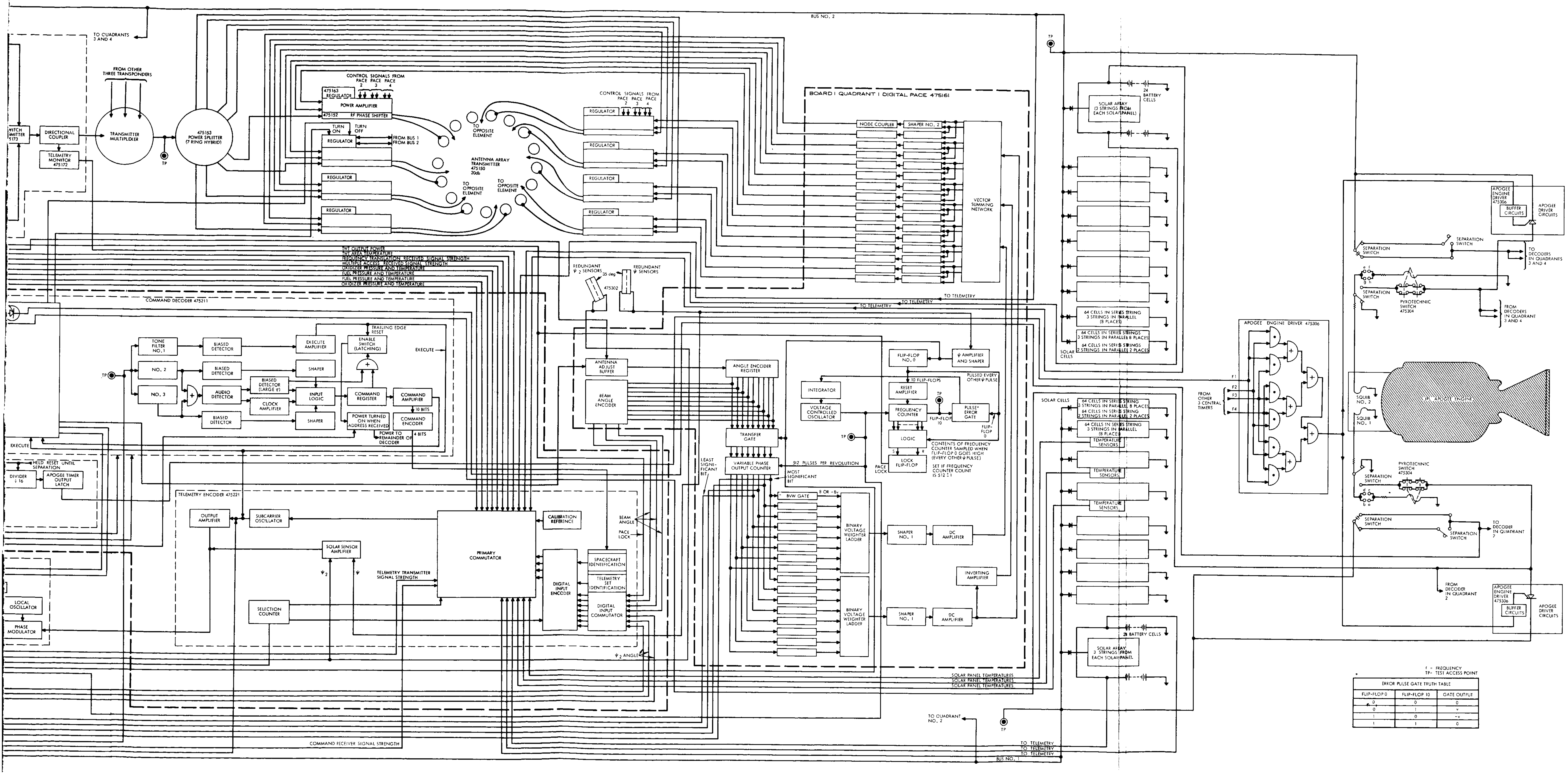
* Certain parameters have been omitted because applicable data were not available at time of publication.



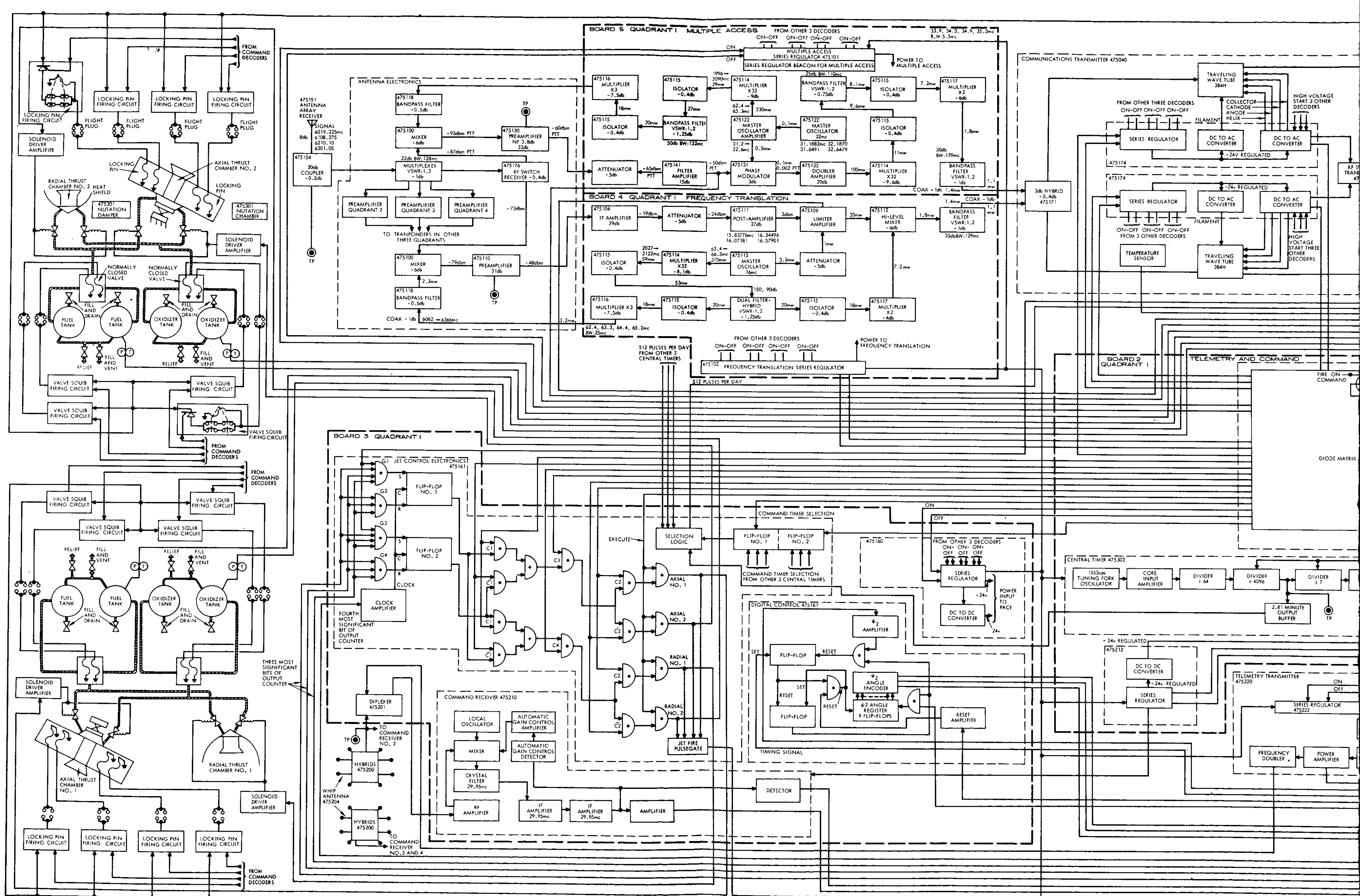
a) Subsystem and major control items

Figure 4-1. Advanced Syncom Block Diagrams



ERROR PULSE GATE TRUTH TABLE

FLIP-FLOP 0	FLIP-FLOP 10	GATE OUTPUT
0	0	0
0	1	+v
1	0	-v
1	1	0



b) System block diagram

Figure 4-1(continued). Advanced Syncom Block Diagrams

TABLE 4-1. TRANSPONDER FREQUENCY ASSIGNMENTS (MC)

Transponder	Input	Output	Beacon	Master Oscillator		IF Frequency Transponder
				Multiple Access	Frequency Transponder	
475025-100	6019.325	3992.09	4006.95	31.1882	15.83776	62.4
475025-101	6108.275	4051.08	4066.16	31.6491	16.01718	63.3
475025-102	6212.10	4119.94	4135.28	32.1870	16.34496	64.4
475025-103	6301.05	4178.93	4194.49	32.6979	16.57901	65.2

3.2.2.5 Frequency Translation Receiver, Peculiar Requirements: The frequency translation receiver shall translate and amplify the signal carrier frequency with no conversion in modulation.

3.2.2.5.1 RF Bandwidth: The 3-db bandwidth for the frequency-translation receiver shall be 25 mc ± 1.5 mc measured between IF input and RF output.

3.2.2.5.2 Received Carrier/Received Noise: The requirements of this specification shall not be imposed unless the received carrier power to the received noise power ratio is at least +20.1 dbw.

3.2.2.5.3 Frequency Stability: The beacon signal frequency shall be stable to within 0.002 percent.

3.2.2.6 Multiple-Access Receiver, Peculiar Requirements: The multiple-access receiver shall convert the single-sideband signals from the IF strip into a phase-modulated signal. This signal shall be multiplied up to the proper microwave frequency and amplified.

3.2.2.6.1 RF Bandwidth: The 3-db bandwidth of the multiple-access receiver shall be 6 mc (+1 mc, -0.5 mc) measured at the preamplifier output.

3.2.2.6.2 Phase Modulator Distortion Noise: The distortion noise generated in the phase modulator shall be so low that it does not represent a limiting factor in meeting CCIR's signal/noise recommendations for noise channels when companders are used.

3.2.2.6.3 Capacity: Each of the multiple-access receivers shall be capable of conveying up to 1200 one-way 4-kc voice channels.

3.2.2.6.4 Test Tone/Fluctuation Noise Ratio: This ratio shall be greater than 47.6 db.

3.2.2.6.5 Test Tone/Intermodulation Noise Ratio: This ratio shall be greater than 50.5 db.

3.2.2.6.6 Test Tone/Noise Ratio: This ratio shall be greater than 45.8 db.

3.2.2.6.7 Inputs-Outputs: The transponder inputs-outputs shall be as given in Table 4-2.

3.2.3 Communications Antenna - 475030: The antenna unit shall receive the incoming 6-gc signals, separate them into the four frequency channels, and supply them to the appropriate receiver. The antenna unit combines the four 4-gc signals from the transmitter unit, processes and transmits them.

3.2.3.1 Receiving Antenna: The receiving antenna shall be capable of receiving 6-gc signals.

3.2.3.1.1 RF Power In: None of the performance requirements of the communication subsystem shall apply unless the input power is at least -106.7 dbw.

3.2.3.1.2 Gain: The antenna gain shall be at least 8 db over the frequency range of 6019.325 to 6301.05 mc.

3.2.3.1.3 Receiving Antenna Pattern Characteristics: The radiation pattern of the receiving antenna shall be omnidirectional in the ϕ-plane and have a minimum beamwidth of 17.3 degrees in all θ planes. The peak of the beam will be at an angle of $\theta = 90$ degrees in all directions of ϕ.

3.2.3.2 Receiver Multiplexer: The receiver multiplexer shall be used to separate the received 6-gc signals into the four frequency channels.

3.2.3.2.1 Input and Output Impedance: The input and output impedance shall be as close as is possible to 50 ohms.

3.2.3.2.2 Frequency: The multiplexer shall be capable of separating the input into four separate frequency channels:

1) 6019.325 mc
2) 6108.275 mc
3) 6212.10 mc
4) 6301.05 mc

3.2.3.2.3 Bandwidth: The bandwidth on the four frequencies listed above shall be ±12.5 mc.

TABLE 4-2. TRANSPONDER - 475025

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
Inputs from:			
475173	1	Multiple access	AM/SSB
475173	1	Frequency translation	WBFM
475211	4	Turn on multiple-access series regulator	Command pulse
475211	4	Turn off multiple-access series regulator	Command pulse
475211	4	Turn on frequency translation series regulator	Command pulse
475211	4	Turn off frequency translation series regulator	Command pulse
Outputs to:			
475171	1	Multiple-access output	PM
475171	1	Frequency translation output	WBFM
475221	4	Multiple-access signal strength	
475221	4	Frequency translation signal strength	

The standard command pulse shall be 60-msec long, have a 5-msec rise time, and be 0.2 volt in amplitude.

3.2.3.2.4 Losses: The maximum loss shall be 1.04 db or less over the frequency range.

3.2.3.2.5 RF Power Input: The power in shall be at least -100.2 dbw.

3.2.3.2.6 Isolation: Isolation between frequency channels shall be at least 17 db at f_o ±44.8 mc (f_o denotes each of the four frequencies listed in 3.2.3.2.2).

3.2.3.3 Transmitting Multiplexer: Transmitting multiplexer shall be used to combine the four 4-gc signals.

3.2.3.3.1 Input and Output Impedance: The input and output impedance shall be as close as possible to 50 ohms.

3.2.3.3.2 Frequency: The transmitting multiplexer shall be capable of accepting frequencies from 3979.59 mc to 4194.49 mc.

3.2.3.3.3 Losses: The losses shall not exceed 0.7 db for the four transponder frequencies as listed in 3.2.3.2.2. The beacon loss shall not exceed 2.0 db.

3.2.3.3.4 RF Power Inputs: The RF power input shall be at least 5.7 dbw.

3.2.3.3.5 Isolation: Isolation between channels shall be at least 17 db at f_o ±44.8 mc (f_o denotes each of the four frequencies listed in 3.2.3.2.2).

3.2.3.4 Phased Array Transmitting Antenna: For the purpose of specifying antenna gain and pattern characteristics, the phased array is defined as consisting of the 16 collinear arrays that make up the radiating portion of the antenna, the transmission lines leading from the outputs of the phase shifters to the element arrays and any matching networks required to obtain a broadband impedance match of the element arrays, the phase shifters, and eight-way power divider.

3.2.3.4.1 RF Power Splitter - 475153: The RF power splitter shall split the RF power into eight equal amplitude and equal phase parts.

3.2.3.4.1.1 Frequency: The frequency range shall be from 3992.09 mc to 4194.49 mc.

3.2.3.4.1.2 Losses: The losses shall be no more than 1 db over the frequency range.

3.2.3.4.1.3 RF Power Input: The RF power input shall be at least 5.0 dbw.

3.2.3.4.1.4 Input and Output Impedance: The input and output impedance shall be made as close as possible to 50 ohms.

3.2.3.4.2 Power Amplifier - 475162: The power amplifiers shall be capable of accepting signals of the four $20 \cos 2\pi (\sin (2\pi\ ft \text{ plus } m\pi/8))$ and $20 \sin 2\pi (\sin (2\pi\ ft \text{ plus } m\pi/8))$.

3.2.3.4.2.1 Gain: The power amplifiers shall have a gain of 2.86 db.

3.2.3.4.2.2 Power Input: The power amplifier shall not require more than ____ amps at -24 volts, ____ amps at 24 volts, ____ amps at -35 volts, and ____ amps at 35 volts.

3.2.3.4.2.3 Power Amplifier Regulators - 475163:

3.2.3.4.2.3.1 Quantity: There shall be a regulator for every power amplifier.

3.2.3.4.2.3.2 Voltage Output: The regulators shall be capable of supplying ±24 volts and ±35 volts.

3.2.3.4.3 Phase Shifter - 475152: The phase shifters shall be capable of inducing 16 different phase shifts (ϕ_n) of $\phi_n = 2\pi \cos(wt + n \cdot 22.5$ degrees), $n = 0, 1, 2, \ldots 15$, where the spacecraft spin rate, ω, shall be $200\pi \pm 100\pi$ radians.

3.2.3.4.3.1 Losses: The maximum losses shall be 1 db over the frequency range.

3.2.3.4.3.2 RF Power Input: The minimum RF power input shall vary over the range 0.4 w/phase shifter to 2 w/phase shifter.

3.2.3.4.3.3 Input and Output Impedance: The input and output impedance shall be made as close as possible to 50 ohms.

3.2.3.4.4 Phased-Array Transmitting Antenna - 475150

3.2.3.4.4.1 RF Power Input: The RF power input shall be at least 3 dbw.

3.2.3.4.4.2 Phased-Array Antenna Gain: The pattern gain, at the peak of the beam, shall be at least 18.0 db over the frequency band from 3992.09 mc to 4194.49 mc.

3.2.3.4.4.3 Phased-Array Antenna Pattern Characteristics: The radiation pattern of the transmitting antenna shall be an elliptically shaped pencil beam a minimum of 17.3 degrees wide in the θ-plane (parallel to the spin axis) and 23 degrees wide in the ϕ-plane (perpendicular to the spin axis). The peak of the beam will be at an angle θ = 90 degrees for any direction of the beam in the ϕ-plane.

3.2.4 Communication Transmitter - 475040: The communication transmitter shall consist of a power amplifier, amplifier power supply, and input-output buffer equipment.

3.2.4.1 Quantity: There shall be four communication transmitters. Each communication transmitter shall consist of:

1)	One 3-db hybrid	475171
2)	One telemetry monitor	475177
3)	Two RF switches	475173 and 475176
4)	Two traveling-wave tubes	384H
5)	Two TWT power supplies	475174

3.2.4.2 3-db Hybrid - 475171: The 3-db hybrid shall be capable of accepting signals from two separate sources and coupling them to either of two separate outputs.

3.2.4.2.1 Losses: The losses including the power split shall not exceed 3.40 db over the frequency range 3992.09 to 4194.49 mc.

3.2.4.2.2 RF Power Input: The RF power input shall be 1.1 ± _____ milliwatt.

3.2.4.2.3 Isolation: Isolation shall be at least 25 db.

3.2.4.2.4 Input and Output Impedance: Input and output impedance shall be made as close as possible to 50 ohms.

3.2.4.2.5 Input-Outputs: 3-db hybrid inputs-outputs shall be as given in Table 4-3.

3.2.4.3 Power Amplifiers: The final power amplifiers shall be traveling-wave tubes 384H.

3.2.4.3.1 Input Power: The input power shall be 1/2 mw ± ____ mw.

3.2.4.3.2 Output Power: The RF output power of each power amplifier shall be at least 4 watts.

3.2.4.3.3 Frequency Band: The TWT shall be capable of operating within specifications over the frequency range 3992.09 mc to 4194.49 mc.

TABLE 4-3. HYBRID - 475171

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
475025	1	Multiple access	PM, 1 mw
475025	1	Frequency translation	WBFM, 1 mw
OUTPUTS TO:			
384H	1	Multiple access	PM, 1/2 mw
384H	1	Frequency translation	WBFM, 1/2 mw

3.2.4.3.4 TWT Electrical Performance: TWT electrical performance shall be found in Table 4-4.

3.2.4.3.5 Inputs-Outputs: TWT inputs-outputs shall be found in Table 4-5.

3.2.4.3.6 Filter: An S-band output filter shall be provided.

3.2.4.4 TWT Power Supply - 475174: The power supply shall provide necessary power for TWT operation.

3.2.4.4.1 Power Input: The power input shall be -24 volts ±1 percent.

3.2.4.4.2 Cathode Voltage: The cathode voltage shall be -1300 volts ±1 percent.

3.2.4.4.3 Helix Voltage: The helix voltage shall be 0 volt.

3.2.4.4.4 Collector Voltage: The collector voltage shall be -725 volts ± 1 percent.

3.2.4.4.5 Anode Voltage: The anode voltages shall be 125 volts ±1 percent.

3.2.4.4.6 High Voltage Start: There shall be a high voltage start pulse. The start pulse shall be a standard command pulse.

TABLE 4-4. ELECTRICAL PERFORMANCE - TWT 384

Frequency	3.9 gc to 4.2 gc
RF power output	4.1 to 4.5 watts
RF saturation gain	40 db
RF small signal gain	50 db
Spurious output (harmonics of operating frequency)	
Noise figure	28 db
Impedance	50
VSWR (input and output)	1.2:1
Maximum load VSWR	Short circuit, any phase
Intermodulation distortion	
Efficiency (excluding heater)	35 percent
Heater power	1.17 watts nominal
Total dc input power	12.1 watts
Cathode voltage	-1300 volts ± ____ percent
Collector voltage	-725 volts ± ____ percent
Collector current	17.6 ma
Helix voltage	0 volt
Helix current	1.7 ma
Anode voltage	125 volts ± ____ percent
Anode current	0
Heater voltage	4.5 volts ± ____ percent
Heater current	0.27 ampere
Predicted life	50,000 hours

Table 4-4 (continued)

Focusing	Platinum - Cobalt magnets Field strength - 750 gauss
Beam transmission with RF	85.5 percent
Cathode	
Base	Ni
Impurities	0.09 percent Zr 0.02 percent Fe 0.001 percent Mn 0.001 percent Si 0.02 percent Cu 0.005 percent W
Cathode loading	0.0854 amp/cm^2
Cathode temperature	720 °C

TABLE 4-5. TRAVELING-WAVE TUBE 384-H

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description	
INPUT FROM:				
475174	1	Cathode	-1300 volts	19.3 ma
475174		Helix	0 volt	1.7 ma
475174	1	Anode	125 volts	0 ma
475174	1	Collector	-725 volts	17.6 ma
475174	1	Filament	4.5	0.27 amp
475171	1	RF	4 gc	1/2 mw
OUTPUT TO:				
475173	1	RF	4 gc	4 watts

3.2.4.4.7 Filament Voltage: The filament power supply shall be -4.5 volts ac ±1 percent.

3.2.4.4.8 Transmitter Regulators: The transmitter regulators shall be a series type.

3.2.4.4.8.1 Power Input: The power input shall be -28 volts unregulated.

3.2.4.4.8.2 Turn-on/Turn-off: The transmitter regulators shall be capable of being turned on and turned off by a standard command pulse from any of the three regulators.

3.2.4.6.1 Quantity: There shall be two RF switches.

3.2.4.6.2 RF Switch, Receiver - 475176

3.2.4.6.2.1 RF Power In: The RF power in shall be at least -100.2 dbw.

3.2.4.6.2.2 Frequency Band: The RF switch receiver shall be capable of operating within specifications over the frequency range of 6019.325 to 6301.05 mc.

3.2.4.6.2.3 Losses: The losses shall not exceed 0.3 db over the frequency range given in 3.2.4.6.2.2.

3.2.4.6.2.4 Switching Current: The switching current shall not exceed 1.0 ampere.

3.2.4.6.2.5 Input and Output Impedance: The input and output impedance shall be made as close as possible to 50 ohms.

3.2.4.6.2.6 Inputs-Outputs: The RF switch input-outputs shall be as given in Table 4-6.

3.2.4.6.3 RF Switch, Transmitter - 475173:

3.2.4.5.3.1 RF Power Input: The RF power input shall be at least 4 watts.

3.2.4.6.3.2 Frequency Band: The RF switch shall be capable of operating within specification over the frequency range of 3992.09 to 4194.49 mc.

3.2.4.6.3.3 Losses: The losses shall not exceed 0.3 db over the frequency range 3992.09 to 4194.49 mc.

3.2.4.6.3.4 Switch Current: The switching current shall not exceed 1.0 ampere.

TABLE 4-6. RF SWITCH

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
Switch, Receiver 475176			
Inputs from:			
Multiplexer		RF	6 gc -1002 dbw
	1	Switch current	1 ampere
Outputs to:			
475025	1	RF multiple access	6019 - 6301 mc
475025	1	RF translation	6010 - 6301 mc
Switch, Transmitter 475173			
Inputs from:			
TWT 384H No. 1	1	RF	4 gc 6 dbw
TWT 384H No. 2	1	RF	4 gc 6 dbw
475174 No. 1	1	Switch	1 ampere
475174 No. 2	1	Switch	1 ampere
Outputs to:			
4715154	1	RF	5. 7 dbw

3.2.4.6.3.5 Input and Output Impedance: The input and output impedance shall be made as close to 50 ohms as possible.

3.2.4.6.4 Input-Outputs: RF switch input-outputs shall be as given in Table 4-6.

3.2.4.7 Telemetry Monitor - 475172: The telemetry monitor shall be capable of monitoring a 4-watt signal for telemetry purposes.

3.3 Antenna and Jet Control Subsystem. The antenna and jet control subsystem is comprised of four sets each of three control items:

1) Phased-Array and Jet Control Electronics, 475035

2) Central Timer, 475303

3) Series Regulator, 475160

This subsystem is responsible for firing the apogee motor at the proper time; developing control signals to provide the spacecraft with the capability of being properly oriented and synchronous in the equatorial plane; and maintaining antenna beam despin rate.

3.3.1 Reliability: The antenna and jet control subsystem shall have a probability of operation within the performance requirements of 0.988 for a 1-year requirement and 0.963 for a 3-year requirement.

3.3.2 Phase-Array and Jet Control Electronics

3.3.2.1 Quantity: There shall be four phased-array and jet control electronics.

3.3.2.2 Phased-Array Control Electronics (PACE): Each PACE shall be able to operate independently of the other three PACE.

3.3.2.2.1 Signals Out: There shall be 16 signals out of the PACE. They shall be $20 \cos 2\pi \; [\sin(2\pi \; ft + m\pi/8)]$ and $20 \sin 2\pi \; [\sin(2\pi \; ft + m\pi/8)]$ where $m = 0, 1, 2, \ldots, 7$.

3.3.2.2.2 Error: The positioning error on the beam shall not exceed ± 1 degree.

3.3.2.3 Jet Control Electronics: The jet control electronics shall be capable of producing four jet fire pulses.

3.3.2.3.1 Command Beam Angle: The command beam angle shall be the angle at which the jets fire. They shall be standard command pulses.

3.3.3 Central Timer - 475303

3.3.3.1 Quantity: There shall be four central timers.

3.3.3.2 Squib Fire Signals: The central timer shall provide a fire signal 315 minutes ±1 percent after separation. Squib fire signal from at least two central timers shall be necessary to fire squibs.

3.3.3.2.1 Output Level: The output level shall be -24 volts.

3.3.3.3 Relative Motion Correction Pulse: The central timer shall provide 512 pulses per day.

3.3.3.3.1 Output Level: The output level shall be -24 volts.

3.3.3.3.3.2 Pulse Width: The pulse shall be 10 microseconds wide.

3.3.4 PACE Inputs-Outputs: PACE inputs-outputs shall be as given in Table 4-7.

3.3.5 Series Regulator - 475160: Each PACE shall have a series regulator for a power supply.

3.3.5.1 Power Out: The regulator shall provide +24 volts at 200 ma and -24 volts at 100 ma. Regulation shall be ±1 percent. Maximum ripple shall be 200 mv peak-to-peak.

3.3.5.2 Turn-on/Turn-off: Each regulator shall be turned on and turned off by separate turn-on/turn-off pulses. The pulses shall be standard command pulses.

3.3.5.3 Failure Turn-Off: Each regulator shall be capable of turning itself off or be capable of being turned off by command in the event of any internal failure.

3.4 Telemetry and Command Subsystem

The telemetry and command subsystem shall provide facilities to receive, process, and execute commands which will control spacecraft operation. It shall also provide facilities to encode digital and analog signals, which indicate quality of operation and transmit these signals to the ground.

3.4.1 Reliability. The telemetry and command subsystem shall have a probability of operation within the performance requirements of 0.99 for a 1-year requirement and 0.98 for a 3-year requirement.

3.4.2 Telemetry and Command Antenna - 475045.

3.4.2.1 Polarization: The polarization of the radiation shall be elliptical. For transmission, the axial ratio of the polarization along the spin axis shall not be greater than 1 db and, as a design objective (not a requirement), the ratio should be less than 3 db to a 30-degree angle with respect to the spin axis. For reception, the axial ratio of the polarization ellipse should be less than 3 db along the spin axis, as a design objective (not a requirement).

3.4.2.1.1 Radiation Pattern: As nearly isotropic a coverage as possible shall be provided. Radiation pattern measurements shall be made with linearly polarized source antennas.

TABLE 4-7. PACE DIGITAL CONTROL - 475161

Control Item	Number of Quadrants	Function	Description
INPUT FROM:			
475211	4	Central timer FF No. 1	Command pulse
475211	4	Central timer FF No. 2	Command pulse
475211	4	Backup command	Command pulse
475303	4	Selection logic	
475211	4	C1	Command pulse
475211	4	C2	Command pulse
475211	4	$\overline{C1}$	Command pulse
475211	4	$\overline{C2}$ Jet fire angle	Command pulse
475211	4	C3	Command pulse
475211	4	C4	Command pulse
475302	4	$\psi 1$	
475302	4	$\psi 2$	
475160	1	Power	+24 volts
475160	1	Power	-24 volts
	4	Timing signal	
475211	4	Pseudo ψ	Command pulse
OUTPUTS TO:			
475221	4	$\psi 2$ angle	
475221	4	$\psi 2$ angle	
475221	4	$\psi 2$	
475221	4	Beam angle	
475221	4	Beam angle	
475221	4	Beam angle	
Axial jet No. 1		Jet command	
Axial jet No. 2		Jet command	
Radial jet No. 1		Jet command	
Radial jet No. 2		Jet command	
475152	4	Phase $n = 0$	
475152	4	shift $n = 1$	
475152	4	drivers $n = 2$	$20 \cos 2\pi$
475152	4	$n = 3$	$[\sin (2\pi ft+m\pi/8)]$
475152	4	$n = 4$	
475152	4	$n = 5$	
475152	4	$n = 6$	
475152	4	$n = 7$	
475152	4	$n = 0$	
			$20 \sin 2\pi$ $[\sin (2\pi ft+m\pi/8)]$

Table 4-7 (continued)

Control Item	Number of Quadrants	Function	Description
475152	4	Phase n = 1	
475152	4	shift n = 2	
475152	4	drivers n = 3	
475152	4	n = 4	
475152	4	n = 5	
475152	4	n = 6	
475152	4	n = 7	
475221	4	PACE lock	

The standard command pulse shall be 60 msec long, have a 5-msec rise time, and be 0.2 volt in amplitude.

3.4.2.1.2 Bandwidth: The center frequency shall be the transmission frequency. Signal strength of the receiving frequency shall not be down more than 3 db.

3.4.2.2 Diplexer

3.4.2.2.1 Quantity: Four diplexers shall be provided.

3.4.2.2.2 Frequency: Two channels shall be provided in each diplexer. One channel shall serve as an input to the receiver, and the other channel shall serve as an output from the transmitter.

3.4.2.2.3 Isolation: The receiver channel shall offer at least 60 db rejection to the transmitter frequency. The transmitter channel shall offer at least 30 db rejection to the receiver frequency.

3.4.2.2.4 Insertion Loss: The insertion loss, with all diplexers, multiplexers and cables installed and properly terminated, shall not exceed 3 db between the input to any transmitter channel and the antenna terminals, or 3 db between the antenna terminal and the input terminals to the receiver.

3.4.2.2.5 Voltage Standing-Wave Ratio: The voltage standing-wave ratio at the receiver and transmitter terminals, with all necessary cabling, multiplexers, diplexers and antennas properly installed on the spacecraft, shall not exceed 1.5:1 at the transmitting frequency and 2.5:1 at the receiving frequency.

3.4.3 Command Group - 475050. The command group shall be composed of

1) Four Command Receivers, 475210

2) Four Command Filter-Decoders, 475211

3) Four Command Regulators, 475212

3.4.3.1 Command Receiver - 475210

3.4.3.1.1 Input Signal: The receiver shall be designed to receive amplitude modulated signals.

3.4.3.1.2 Frequency: The center of the frequency pass band shall be ____ mc.

3.4.3.1.3 Noise Figure: The noise figure of the receiver shall be 10 db maximum referenced to the standard source temperature of 290°K.

3.4.3.1.4 Receiver Stability: The receiver shall remain within 0.003 percent of the selected frequency.

3.4.3.1.5 IF Bandwidth: The bandwidth at points 3 db down from maximum response shall be 60 kc ±15 kc.

3.4.3.1.6 Input and Output Impedance: The input and output impedance shall be 50 ohms.

3.4.3.1.7 Sensitivity: For amplitude modulated input signal levels of -95 dbm or greater the receiver output shall be sufficient to operate all command functions.

3.4.3.1.8 Input-Output: Command receiver input-outputs shall be as given in Table 4-8.

3.4.3.2 Command Decoder - 475211

3.4.3.2.1 Operation: The command decoder shall process the audio signal so as to generate command signals to all required circuitry.

3.4.3.2.2 Command Pulse Duty Cycle: The command receivers and that portion of the decoder circuitry required to initiate full command turn-on shall be operating at all times that spacecraft power is on.

3.4.3.2.3 Redundancy: The command system shall be interconnected so that failure of one of the multiple systems does not compromise the ability of remaining systems to perform all command functions.

3.4.3.2.4 Real-Time Operation: The design of the command system shall be predicated on the necessity of a real-time RF link for the execute signal.

3.4.3.2.5 Inputs-Outputs: Command decoder inputs-outputs shall be as given in Table 4-9.

TABLE 4-8. COMMAND RECEIVER - 475210

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
Whip antenna		Command receiver	148 mc
475212	1	Power supply	-24 volts
OUTPUTS TO:			
475221	4	AGC	
475211	1	0, 1, execute tones	Awaiting NASA frequency determination

TABLE 4-9. COMMAND DECODER - 475211

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
475210	1	0, 1, execute tones	Awaiting NASA frequency determination
475212	1	Power supply	±24 volts regulated
475212	1	Power supply	-24 volts unregulated
OUTPUTS TO:			
475160	4*	Turn on series regulator	Command pulse
475160	4*	Turn off series regulator	Command pulse
475025	4*	Turn on multiple-access series regulator	Command pulse

Table 4-9 (continued)

Control Item	Number of Quadrants	Function	Description
475025	4*	Turn off multiple-access series regulator	Command pulse
475025	4*	Turn on frequency translation series regulator	Command pulse
475025	4*	Turn off frequency translation series regulator	Command pulse
475175	4*	Turn on series regulator for TWT No. 1	Command pulse
475175	4*	Turn off series regulator for TWT No. 1	Command pulse
475175	4*	Turn on series regulator for TWT No. 2	Command pulse
475175	4*	Turn off series regulator for TWT No. 2	Command pulse
475160	4	Central timer select FF No. 1	Command pulse
475160	4	Pseudo ψ	Command pulse
475160	4	Central time select FF No. 2	Command pulse
	4	Jet fire backup command	Command pulse
	1	C1	Command pulse
	1	$\overline{\text{C1}}$ Jet fire angle	Command pulse

Table 4-9 (continued)

Control Item	Number of Quadrants	Function	Description
475160	1	C2	Command pulse
	1	$\overline{C2}$	Command pulse
	1	C3	Command pulse
	1	C4	Command pulse
475174	4*	High voltage start TWT No. 1	Command pulse
	4*	High voltage start TWT No. 2	Command pulse
475160	4	Command antenna beam angle	Command pulse
475221	4	Execute command	Command pulse
475221	4	Encoded command register	
475221	4	Encoded command register	
475221	4	Encoded command register	
475221	4*	Turn on	Command pulse
	4*	Turn off	Command pulse

* One distinct and different signal shall be sent to each of the quadrants. The standard command pulse shall be 60 msec long, have a 5-msec rise time, and be 0.2 volt in amplitude.

3.4.3.3 Command Regulator - 475212: The command regulator shall provide -24 volts ±1 percent to both the command receiver and command decoder. It shall also provide +24 volts ±1 percent to the command decoder.

3.4.4 Telemetry Group - 475055. The telemetry group shall consist of:

1) Four Telemetry Transmitters, 475220

2) Four Telemetry Encoders, 475221

3) Four Telemetry Regulators, 475222

3.4.4.1 Telemetry Transmitter - 475200

3.4.4.1.1 Frequency: Two of the transmitters shall be designed to operate at a frequency of ____ mc and the other two transmitters should be designed to operate at a frequency of ____ mc.

3.4.4.1.2 Frequency Stability: The transmitter frequency shall remain within 0.003 percent under normal service conditions.

3.4.4.1.3 Power Output: The transmitter output shall be at least 1 watt into a 50-ohm load.

3.4.4.1.4 Modulation: The modulator portion of the transmitter shall angle-modulate the RF signal generated in the transmitter.

3.4.4.1.5 Modulation Sensitivity: The modulation sensitivity shall provide 1.5 radians phase deviation for normal input from the encoder.

3.4.4.1.6 Input and Output Impedance: Input and output impedance shall be 50 ohms.

3.4.4.1.7 Inputs - Outputs: The telemetry transmitter inputs-outputs shall be as given in Table 4-10.

3.4.4.2 Telemetry Encoder - 475221: The telemetry encoder shall be capable of commutating the input signals (Table 4-10) and providing a suitable modulation signal to the telemetry transmitter.

3.4.4.2.3 Input and Output: The telemetry encoder input-outputs shall be as given in Table 4-11.

3.4.4.3 Telemetry Regulator - 475200: The telemetry regulator shall provide -24 volts ±1 percent. It shall be a series-type regulator.

3.4.4.3.1 Turn-on/Turn-off: The telemetry regulator shall be capable of being turned on and turned off by a standard command pulse from any of the four command decoders.

TABLE 4-10. TELEMETRY TRANSMITTER - 475220

INPUTS FROM:	
Same as outputs from 475221 and -24 volt power supply from 475222	
OUTPUTS TO:	
475201	Same as inputs except for -24 volts

TABLE 4-11. TELEMETRY ENCODER - 475221

Inputs and Outputs

Control Item	Number of Quadrants	Function	Description
INPUTS FROM:			
475061	4	Jet fire	Digital
475211	4	Execute	Digital
Battery No. 1		Unregulated bus voltage No. 1	Analog
Battery No. 2		Unregulated bus voltage No. 2	Analog
475252		Solar panel temperature	Analog
475161	4	PACE lock	Digital
475161	4	Antenna beam angle	Digital
475161	4	Antenna beam angle	Digital
475161	4	Antenna beam angle	Digital
475211	4	Command verification	Digital

Table 4-11 (continued)

Control Item	Number of Quadrants	Function	Description
475211	4	Command verification	Digital
475211	4	Command verification	Digital
475302	4	ψ_2 angle	Digital
475302	4	ψ_2 angle	Digital
475302	4	ψ_2 angle	Digital
	4	Propellant tank pressure No. 1	Analog
	4	Propellant tank pressure No. 2	Analog
	4	Propellant tank pressure No. 3	Analog
	4	Propellant tank pressure No. 4	Analog
475172		Transmitter power No. 1	Analog
475172		Transmitter power No. 2	Analog
475172		Transmitter power No. 3	Analog
475172		Transmitter power No. 4	Analog
475025		Receiver signal strength No. 1	Analog
475025		Receiver signal strength No. 2	Analog
475025		Receiver signal strength No. 3	Analog
475025		Receiver signal strength No. 4	Analog
47522	1	Power supply	-24 volts

Table 4-11 (continued)

Control Item	Number of Quadrants	Function	Description
475302	4	ψ	Analog
475302	4	ψ_2	Analog
OUTPUTS TO:			
475161	4	Timing signal	
475220	Same as inputs excluding -24 volts (power supply) and including spacecraft identification, T/M set identification, and calibration reference.		

3.4.5 General Subsystem Requirements

3.4.5.1 Countdown Testing: The beacon tracking, telemetry, and command subsystem shall operate with the nose fairing in place to the extent necessary to control and measure proper performance of the spacecraft.

3.4.5.2 Operating Duty Cycle: The telemetry and command subsystem shall be designed to have the capability of continuous operation.

3.4.5.3 Telemetry and Test Signal Provisions: Test plugs shall be provided on the units of the telemetry and command subsystem as required to allow appropriate signal voltages to be monitored for telemetry and test purposes.

3.5 Power Supply Subsystem - 475060: The electrical power subsystem shall provide the spacecraft electronics with operating power. The electrical power will be provided by solar panels. A battery shall be provided to supply power during eclipses and peak loads. The subsystem shall consist of

1) 96 battery cells

2) 16 solar panels

3.5.1 Reliability. The power subsystem shall have a probability of operation within the performance requirements of 0.998 for a 1-year requirement and 0.994 for a 3-year requirement.

3.5.2 Solar Array

3.5.2.1 Quantity: There shall be sixteen solar panels. They shall consist of fourteen Solar Panels - 475252 and two Solar Panels, Special - 475253.

3.5.2.2 Common Requirements

3.5.2.2.1 Solar Array Output: The solar array output shall be 28.0 volts and 5.14 amperes, at a solar intensity of 140 $mw/cm,^2$ a temperature of 77°F and a sun incidence angle of 90 degrees ±25 degrees.

3.5.2.3 Peculiar Requirements

3.5.2.3.1 Solar Panel, Special - 475253: The special solar panels shall be similar to the Solar Panel - 457252, having a different cell layout to accommodate the radial jet.

3.5.3 Battery

3.5.3.1 Quantity: There shall be 96 nickel-cadmium battery cells.

3.5.3.2 Discharge-Charge Efficiency: The discharge-charge efficiency is defined as the ratio of the ampere-hours removed from a fully charged battery during discharge to the ampere-hours required to restore it to its originally fully charged condition. For the initial system design, the discharge-charge efficiency shall be greater than 36 percent (including change regulations).

3.5.3.3 High Current Discharge: The cell terminal voltage during discharge of a fully-charged cell at a load of 12.0 amperes shall be 1.0 volt minimum for a period of 10 seconds.

3.5.3.4 Capacity at High Rate Discharge: The cell discharge capacity at 75°F shall be a minimum of 4.8 ampere-hours when discharged at a constant current of 6.0 amperes to an end voltage of 1.0 volt.

3.5.3.5 Power Capacity: The capacity shall be:

6.0 amp-hr at 75°F at 1.2 amperes discharge rate

4.8 amp-hr at 100°F at 1.2 amperes discharge rate

4.8 amp-hr at 30°F at 1.2 amperes discharge rate

3.5.3.6 Maximum Charge Rate: The maximum charge rate shall be 5 amperes. The maximum overcharge rate shall be 0.6 ampere continuous.

3.5.3.7 Maximum Charge Voltage: The maximum charge voltage shall be 1.48 volts.

3.5.3.8 Cycle Life: The minimum cycle life shall be 10,000 at 25 percent depth.

3.5.4 Battery Charge Current. Three solar cell strings on each of four solar panels shall provide charging current to 24 battery cells. The other three sets of 24 battery cells shall be charged in the same manner.

3.5.5 Power Distribution. The power shall be distributed to the system by two separate buses as shown in Table 4-12.

3.5.5.1 Power Distribution per Bus: Each bus shall provide power to two quadrants, one-half of the reaction subsystem, phase shifter power amplifiers, and one of the apogee motor squib firing circuits.

3.5.5.2 Bus Source: Each bus shall obtain its power from eight solar panels and 48 battery cells.

3.5.6 Unregulated Bus Characteristics

3.5.6.1 Voltage: The unregulated bus voltage shall be within the limits of -27 and -36 volts dc during normal operating conditions in orbit with all equipment functioning properly. Necessary power regulation or conversion shall be provided as part of each subsystem.

3.5.6.2 Transients: During transfer of the spacecraft equipment loads from any mode of operation to another the voltage at the equipment terminals shall remain within the range of -24 to -37 volts dc and shall recover and remain within the steady-state limits in less than 0.5 second.

3.5.6.3 Ripple: The peak-to-peak ripple voltage output of the solar panels, measured on the unregulated bus, shall not exceed 0.5 volt. The ripple frequency shall be less than 1000 cps.

3.5.7 Discharge Control. Discharge control shall be provided by dropout of the loads under reduced voltage input. The dropout voltage shall increase when loads over the rated value are placed on a subsystem regulator by defective circuitry. The removal of loads by regulator dropout shall not impair later normal use of the subsystem when the voltage is restored to normal values.

3.6 Structure Subsystem

3.6.1 Structure. The spacecraft structure shall provide the basic support for the other subsystems of the spacecraft and for the attachment to the spacecraft support structure of the boost vehicle.

TABLE 4-12. POWER DISTRIBUTION PER BUS

Item	Number of Units per Quadrant	Milli-amperes per Unit	Number of Units per Bus	Bus 1 Units Operating	Bus 1 Load MA	Bus 2 Units Operating	Bus 2 Load MA
Telemetry transmitter	1	245	2	1	245		
Encoder	1	27	2	1	27		
Command receiver	1	27	2	2	54	2	54
Digital electronics (PACE)	1	30	1	1	30		
Phase shifters and power amplifiers	1 unit common to both buses	620	1/2	1/2	310	1/2	310
Communication receivers	1	75	2	2	150	2	150
TWT (4-watt)	2	693	2	2	1386	2	1386
TOTAL					2202*		1900*

* Typical totals. Either bus shall have to carry up to 2512 ma.

3.6.1.1 Accessibility: Accessibility and the capability of quick removal of components shall be considered in the design for mounting subsystems. The covers of the spacecraft shall be removable to permit maximum accessibility to the internal subsystems for replacement, repair, and checkout with minimum weight expenditure.

3.6.1.2 Appendages: Four tracking, telemetering, and command antennas shall be mounted on the forward (direction of launch or +Z) end of the spacecraft. The communications antenna shall be mounted on the aft (or -Z) end of the spacecraft on the spin axis.

3.6.1.3 Use of Shock and Vibration Isolators: The use of shock and vibration isolators shall require approval of the GSFC Project Manager.

3.6.2 Thermal Control Requirements. Thermal control of the spacecraft shall be accomplished by passive and/or active temperature design of the structure. Temperatures shall be controlled on each spacecraft part, unit, or subsystem within a range compatible with its function and its reliability requirements.

3.6.3 Reliability. The structure subsystem shall have a probability of operation, within the performance requirements of 0.997 for a 1-year requirement and 0.991 for a 3-year requirement.

3.6.4 Nutation Damper - 475301. The spacecraft shall utilize two nutation dampers to dissipate the spacecraft nutation energy as heat. The design shall limit the maximum nutation angle to 1 degree when the spacecraft is processed 135 degrees. The time constant of this device shall be less than 1 hour.

3.6.5 Spin Rate Control. Provision shall be made for controlling the spin rate to within 100 ±50 rpm.

3.6.6 Sun Sensor Assembly - 475302. Each sun sensor assembly shall provide a means of determining the angle between the spin axis and the sunline as well as synchronization data on the spacecraft spin rate.

3.6.6.1 General Description: Each sun sensor assembly shall consist of two ψ sun sensors and two ψ_2 sun sensors. The inner ψ and inner ψ_2 are alignment references. The outputs of these sensors will be utilized by the onboard electronics. The outputs of the outer ψ and outer ψ_2 will be telemetered. The beam plane of each ψ_2 sensor shall be at an angle of 35 ±0.5 degree with respect to its reference ψ sensor. The sun sensor assembly, in conjunction with on-board circuitry, shall be capable of measuring the angle between the spin axis and the sunline, ϕ_S, within 1 degree when the angle is within 90 ±25 degrees.

3.6.6.2 Installation and Alignment: Four sun sensor assemblies shall be used on each spacecraft. The sun assemblies will be placed around the circumference of the spacecraft at 0, 90, 180, and 270 degrees ±0.1 degree, relative to the reference radial jet. The beam plane of the reference ψ sensor will be parallel to the spin axis within ±0.25 degree. The beam plane of the reference ψ sensor will be parallel to a radial line of the spacecraft within ±0.1 degree. The line formed by the intersection of the ψ and ψ_2 planes will be perpendicular to the spin axis within ±___ degree.

3.6.6.3 <u>Output Signal Characteristics</u>: A sensor will provide an output signal when the sunline and its beam plane coincide.

3.6.6.3.1 <u>ψ Sensors</u>

3.6.6.3.1.1 <u>Output Voltage</u>: The sensor output voltage shall be between the limits shown in Figure 4-2 when the sunline and its beam plane coincide.

3.6.6.3.1.2 <u>3-db Beam-Width</u>: The sensor 3-db beam width shall be $\left[\frac{0.8° \pm 12.5 \text{ percent}}{\sin \phi_s}\right]$ for ϕ_s values from 30 to 150 degrees.

3.6.6.3.1.3 <u>Sensor Triggering Level</u>: The sensor shall have an output of ___ mv at an angle of $\left[\frac{\theta_T \pm \text{\_\_\_ percent}}{\sin \phi_s}\right]$ from the center of its beam plane for ϕ_s values from 30 to 150 degrees.

3.6.6.3.2 <u>ψ_2 Sensors</u>

3.6.6.3.2.1 <u>Output Voltage</u>: The sensor output voltage shall be between the limits shown in Figure 4-2 when the sunline and its beam plane coincide.

3.6.6.3.2.2 <u>3-db Beam-Width</u>: The sensor 3-db beam-width shall be ___ for ϕ_s values from 35 to 145 degrees.

3.6.6.3.2.3 Sensor Triggering Level: The sensor shall have an output of ___ mv at an angle of $\left[\frac{\theta_T \pm \text{\_\_\_ percent}}{\sin \phi_s}\right]$ from the center of its beam plane for ϕ_s values from 35 to 145 degrees.

3.6.7 <u>Pyrotechnic Switch Assembly</u>. Two pyrotechnic switch assemblies are used to ensure that an open circuit exists between the unregulated bus and the apogee motor squibs after the squibs have been fired. The requirements for this assembly are contained in the Procurement Specification.

3.6.8 <u>Separation Switch</u>. Four separation switches shall be provided to define by telemetry separation of the spacecraft from the Agena and to provide power to the central timer. The requirements for the item are contained in the Procurement Specification.

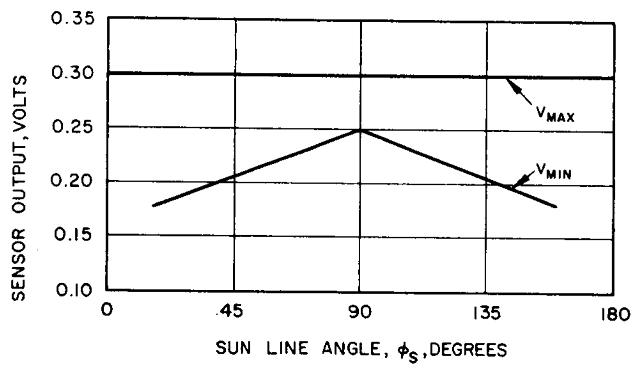


Figure 4-2. Sensor Output Voltage Limits

3.7 Wire Harness Subsystem - 475300

3.7.1 Wire

3.7.1.1 Voltage Drop: The electrical power and signal distribution system shall be so designed that at no time shall any terminal voltage fall below the rated value due to excessive voltage drop in response to transmission of rated currents.

3.7.1.2 Mechanical Strength: No wires smaller than size AWG-26 or equivalent etched circuit lines shall be used in the electrical power and signal distribution system.

3.7.1.3 Installation of Wiring: The installation of wiring shall be in general accordance with the applicable requirements of specification MIL-W-8160.

3.7.2 Harness Construction. The harness shall be constructed to minimize noise effects and to minimize physical damage from heat, flexing, and centrifugal force.

3.7.3 Twisting. Twisting shall be used whenever necessary to eliminate noise effects.

3.7.4 Shielded Wire. As a general rule, there shall be no electrical connections between the shield of any shielded wire and any electrical circuitry. As a general rule, the shield of a shielded wire shall be grounded only at one end.

3.7.5 Grounding

3.7.5.1 Common Ground: System return leads requiring grounding shall be terminated as close as possible to the positive battery ground terminal.

3.7.5.2 Unit Case Grounds: A component case may be considered a shield. One connector pin on each unit may be electrically connected to the unit case internally. The case may be grounded by mechanically contacting a system ground plane or by wiring to the case ground connector pin.

3.7.6 Soldering. All soldering operations that are to be made on Syncom II must be done by personnel that have been certified as passing the requirements set forth in NASA Document MSFC-PROC-158B, "Procedure for Soldering of Electrical Connectors," (High Reliability) 15 February 1963.

The following numbers indicate paragraph numbers of the referenced specification to which exception shall be taken.

VI A 2c — If tubing cannot be used at abrasion points, wrapping may be substituted. If tubing cannot be fitted over a terminal, it may be dispensed with if the terminal has been correctly soldered, and sufficient clearance exists. MIL-I-22129 is waived.

VI A 1d — Preparation of new terminals should not be necessary. Terminals that are being reworked may be cleaned as necessary, care being taken not to damage the terminal or surrounding area.

VI A 1g — Silver-plated wire presently utilized on Syncom II is insulated with fluorinated ethylene propylen (FEP) and is acceptable.

VI A 2a — Any thermal stripper, such as American Missile Products MOD. WS-17B, Ideal Model No. 45-130 or 45-141, that, in the opinion of the Quality Control inspector, does a satisfactory job, is authorized.

VI A 2d — Heat sink tools will continue to be used. However, the circumferential edges of the tool will be rounded and burnished to reduce the possibility of wire abrasion.

VI A 2h — Resistance soldering will not be used until it has been thoroughly tested and established as a satisfactory technique.

VI A 3b — The environmental conditions described cannot at present be met. However, all good housekeeping procedures will be followed, as described in VI A 3a.

To reduce the possibility of accidental damage from loose tools, soldering technicians will use tool trays.

Harness boards will be cleaned regularly to avoid inclusions of debris in the harness assemblies.

VI A 3e(2) — The exposed wire between pot and insulation shall be limited to 0.1 inch. If by chance this tolerance is exceeded, NASA will consider and may approve the condition on an individual basis.

VI B 1d — The solder should follow the cup contour as closely as is reasonable under the particular circumstance and should have a slightly concave appearance. However, a convex configuration is acceptable provided the solder does not protrude beyond the outer diameter of the pot. It is recognized that the soldering process may leave a small amount of solder adhering to the outer surface of the pot, but excessive amounts, in the opinion of the Quality Control inspector, shall not be allowed. Inspections will be made with five to ten power magnifiers. In cases of controversy a higher power magnification may be used.

VI B 2a — Where design or layout necessitates the connecting of more wires than the terminal was designed for, the extra wires may be wrapped around the terminal. All caution will be used to ensure proper clearance, insulation, and correct soldering. A sufficient excess length of wire may be used on terminal posts to permit one wiring change. These connections shall be heavily coated with protective paint to provide support as well as insulation.

VI B 1f — Wicking should not extend beyond 0.25 inch from the solder pot and connections which exceed this limit will be rejected if the insulation is bulged.

VI B 3 — Where necessary, in the opinion of the Quality Control inspector, the wrap may be increased up to 270°.

3.7.7 Filtering. RF filters shall be provided on power line inputs to RF circuitry.

3.8 Apogee Motor Subsystem. This subsystem shall consist of a solid propellant rocket engine to be used for injection of the Syncom II spacecraft into a nominally circular equatorial orbit from the apogee of a transfer orbit. This subsystem is GFE. The requirements for this subsystem are defined in Procurement Specification for Syncom II Apogee Rocket Motor; Buyer: National Aeronautics and Space Administration; Contractor: Jet Propulsion Laboratory.

3.8.1 Apogee Motor Drive Circuit. The apogee motor drive circuit shall provide an output of 15 ±5 amperes for ±10 msec.

3.8.1.1 Quantity: There shall be two apogee motor drive circuits.

3.8.1.2 Output Impedance: The output impedance shall be 4 ± ohms.

3.8.1.3 Reliability: The reliability of the apogee motor drive circuit shall be at least 0.999.

3.9 Reaction Control Subsystem

The reaction control subsystem shall provide two redundant sources of thrust to correct spacecraft longitude, provide the spacecraft with the capability of being synchronous in the equatorial plane, maintain spacecraft spin rate, and orient the spacecraft so that the antennas will illuminate the earth continuously. The unit shall consist of storage tanks for fuel and oxidizer, injector solenoid valves, fuel and oxidizer lines, and thrust chambers aligned axially and radially. The requirements for this subsystem are defined in Hughes Specification X-254044, Procurement Specification for Syncom II Bipropellant Reaction Control System.

3.9.1 Axial Jet Locking Pin Firing Circuit. The axial jet locking pin firing circuit shall provide an output of 15 ±5 amperes for 10 ±2 msec upon command.

3.9.1.1 Reliability: The reliability of the axial jet locking pin firing circuit shall be at least 0.999.

3.9.1.2 Quantity: There shall be eight axial jet locking pin firing circuits.

3.9.1.3 Output Impedance: The output impedance shall be 4 ± ohms.

3.9.2 Valve Squib Firing Circuit. The valve squib firing circuit shall provide an output of 15 ±5 amperes for 10 ±2 msec upon command.

3.9.2.1 Quantity: There shall be eight valve squib firing circuits.

3.9.2.2 Output Impedance: The output impedance shall be 4 ± ohms.

3.9.2.3 Reliability: The reliability of the valve squib firing circuit shall be at least 0.999.

COMMUNICATION TRANSPONDER

General Discussion

The major changes in the transponder block diagram (Figure 4-3) ocurring in this report period are in relation to the varactor high-level mixer. Tests of this unit indicate the need for 4-gc isolators, a new item to be developed, preceding and following the mixer.

Quadrant packaging design is now approximately 40 percent complete. The multiple-access portion is about 80 percent designed, and the frequency-translation part is about 20 percent designed. The structural design has been completely determined.

A preliminary analysis was made of the AM feedback path through the frequency translator transponder. The values shown in Figure 4-4 for the loss, gain, and/or isolation of the various units are estimated. The gain margin found is 125 db. This figure is excluding possible coupling through the dc supply and leakage of connectors and coaxial cables.

Transponder Components Status

Circuits Common to Frequency-Translation and Multiple-Access Transponders

Ferrite Switch, 6 gc. Parameters have been determined from tests in a breadboard model. Design for a final model has been determined and is ready for fabrication.

Ferrite Switch, 4 gc. One final design type is being fabricated. The RF circuit is determined completely, but the external magnetic circuit will be modified from the previous model to accommodate the switch operating circuits in the transponder receiver power supplied.

Hybrid, 3 db. The final design test model is complete and ready for test It is a lightweight model containing OSM-type connectors.

Isolators, 2 gc. One for each frequency of the final type design is being fabricated.

Mixer, 6 gc. Fabrication of parts for eight units is complete.

Receiver Coupler, 20 db, 6 gc. Two ground planes are complete, and stripline cards are being made.

Dual-Coupler Detector. Ground planes are completed. Photo-etched card is in fabrication. Detector mounts are due from the vendor during the next reporting period.

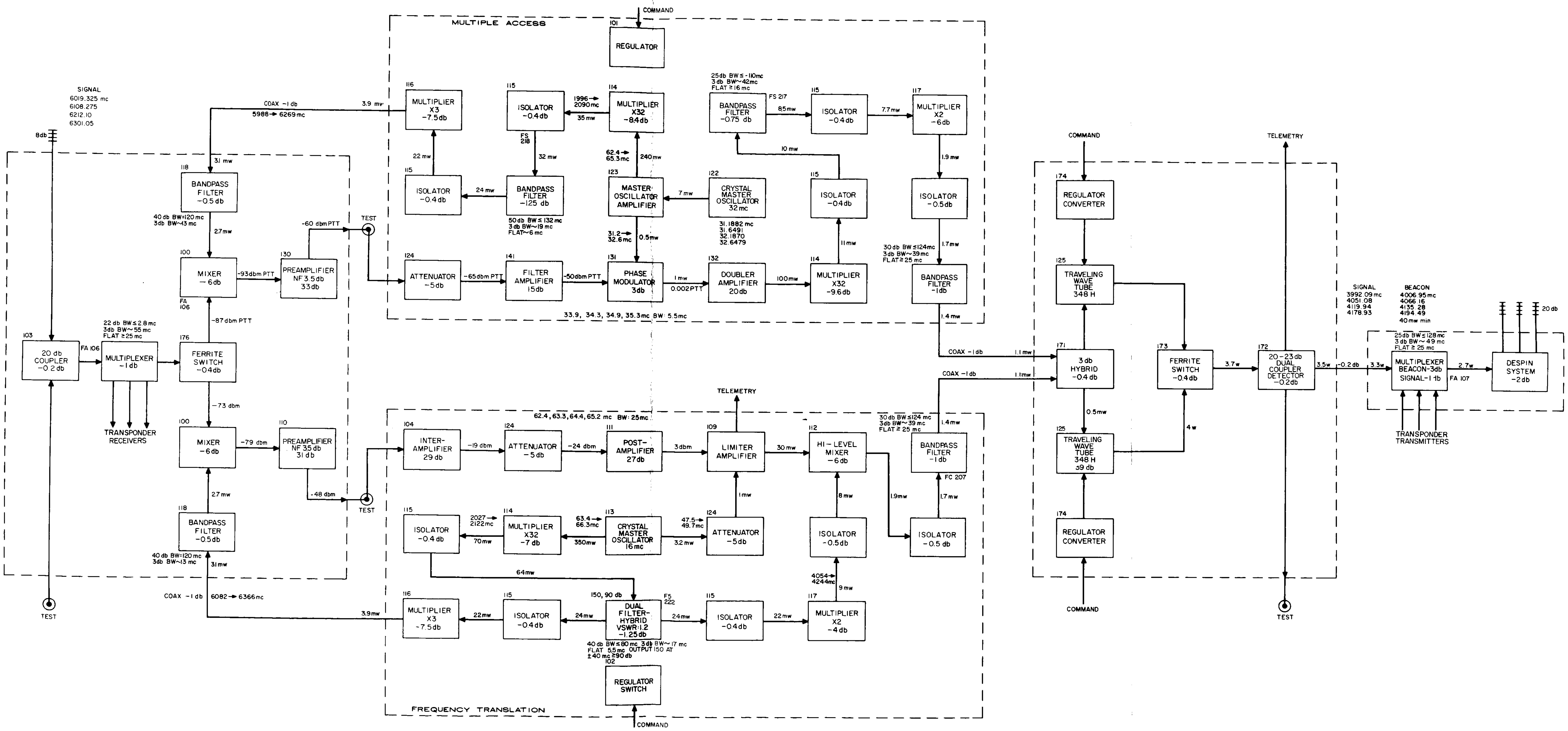


Figure 4-3. Syncom II Transponder

X3 Multiplier. Drawings are complete and parts are being manufactured. Figure 4-5 shows the prototype unit.

X32 Multiplier. The fourth and fifth stages are being modified to extend the frequency range. The fifth stage is mounted on its own base as a unit, and output loop tuning has been incorporated. The Johansen tuning capacitors in the fourth stage have been deleted. Small coaxial tuning capacitors in the inputs of both stages are to be added. Preliminary checks of the fourth and fifth stages as a pair show sufficient bandwidth and power range for all purposes required. The layout of the entire X32 multiplier has been modified to accept approved feedthrough capacitors.

Bandpass Filter, 4 gc. Units are due from Rantec on 15 August.

Receiver Multiplexer. Insertion loss specification at center frequency ± 12.5 mc has been reduced from 1.0 to 0.8 db maximum. The unit is due from Rantec on 4 October. Mountings are still to be determined.

Transmitter Multiplexer. The order to Rantec has been reduced from two units to one. The specification for rejection at ± 64 mc from the center frequency has been reduced from 30 to 25 db minimum.

Circuits for Multiple-Access Transponder

Master Oscillator Doubler Amplifier. Circuit design is complete, and packaging is in process.

Master Oscillator. Thermal design remains to be confirmed. Present plans are for a 2-inch cube of stainless steel containing circuit surrounded with 1-inch-thick foam. A vacuum chamber for testing this unit has been ordered.

Preamplifier. Circuit design is complete. Measured noise figure of the breadboard is 3.5 db, and gain is 35 db.

Narrow- and Wide-Band Filters, 2 gc. Units are due from Rantec on 15 August.

Circuits for Frequency-Translation Transponder

Preamplifier, Intermediate Amplifier, and Postamplifier. Circuit design is complete on these units. Measured characteristics are:

Noise figure at 60 mc:	3.8 db
Output power at onset of limiting:	3.8 mw.
Overall gain exclusive of attenuator:	87 db.

Limiter Amplifier. Modification of the output circuit to operate with the varactor high-level mixer is complete. Work is in process for reducing the unit power consumption consistent with the lower than expected power requirements of the high-level mixer.

High-Level Mixer. The breadboard model has been tested with inputs of 7.2 mw at 4184 mc and with 7.4 mw of available 64-mc power from a 50-ohm source to obtain 1.4 mw at 4120 mc at the output of the following filter. Tests indicate the need for input and output 4 gc isolators to prevent the output from being critically dependent on source and load impedances. The IF impedance at 64 mc measured through two 9-inch 50-ohm cables is 600 ohms resistance in parallel with 70 micromicrofarad. Satisfactory operation is obtained without bias voltage, i.e., with a dc short across the diodes, simplifying the driving circuits.

Isolator 4 gc. Electrical design has been determined. A final type model is being fabricated.

Dual-Filter Hybrid. Units are due from Rantec on 15 August.

Transponder Regulators

Multiple-Access Mode Regulator (101 Unit)

It was estimated during the previous report period that the 101 unit (Figure 4-6) would have ± 0.25 percent static load, line, and temperature regulation. Regulation values in this and subsequent reports will be presented in terms of total deviation (actual ΔV or percentage). This deviation will not necessarily be symmetrical about the nominal value.

According to this method of specifying regulation, the ± 0.25 percent regulation previously estimated would simply be a 0.5 percent total deviation at approximately -24 volts. Test data are now available to substantiate this estimate. One breadboard showed a total deviation of 0.2 percent (ΔV = 48 mv). However, an adverse tolerance accumulation might result in load, line, and temperature regulation slightly in excess of 0.5 percent. Table 4-14 provides a breakdown of line, load, and temperature regulation.

It should be noted that the values in Table 4-14 represent static regulation, i.e., regulation against relatively slow changes in load resistance, unregulated input voltage, and ambient temperature. The dynamic response to higher frequency pertubations will be determined later. The regulation is given in terms of actual voltage change and in percent of -24 volts. Also, both measured values and worst case estimates are given. However, the term "worst case" must be clarified. Time has thus far not permitted a classical worst case analysis of this regulator. What is listed as worst case is an intuitive estimate based upon Syncom I experience plus data involving limited substitution of the most sensitive components.

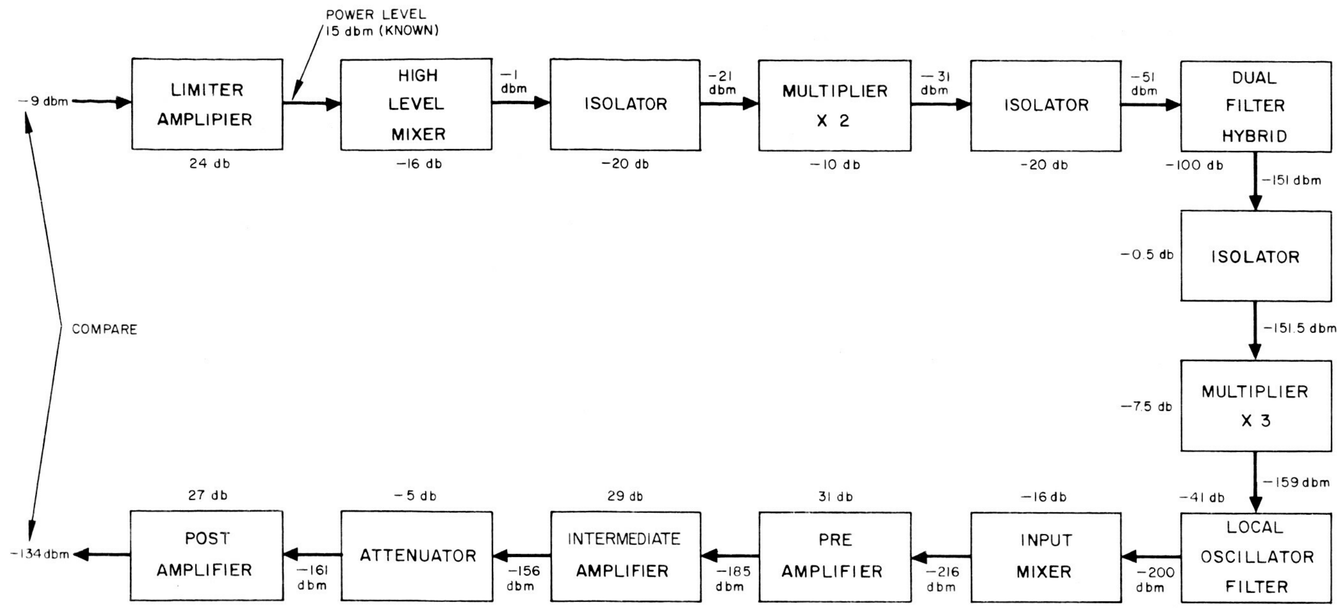


Figure 4-4. Frequency Translation Transponder, AM Feedback Loop

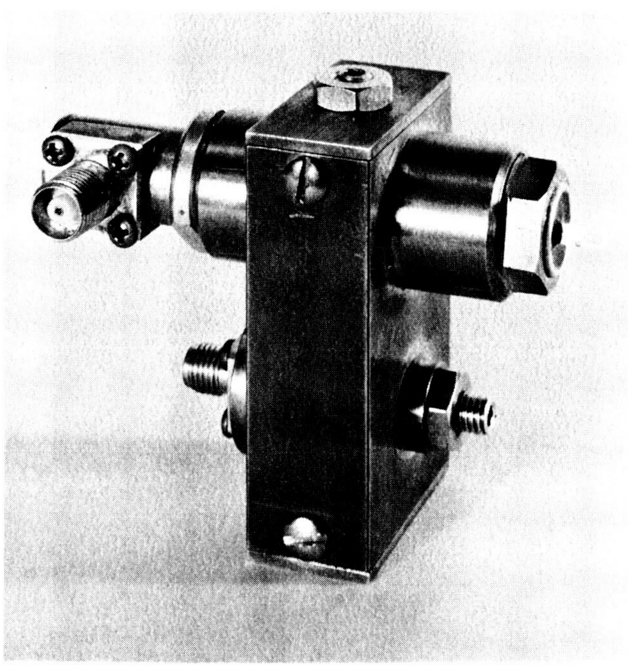


Figure 4-5. Prototype X3 Multiplier

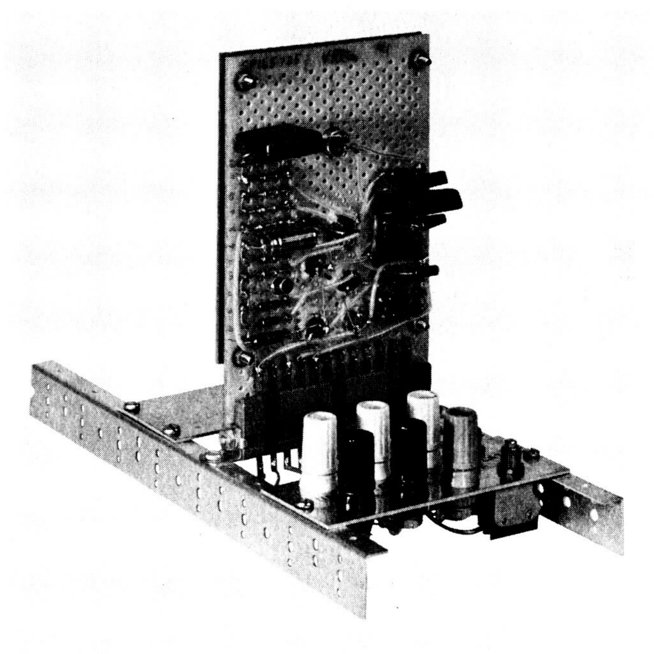


Figure 4-6. Breadboard of Transponder Regulator, MA Mode

TABLE 4-14. VOLTAGE REGULATION, MULTIPLE-ACCESS MODE REGULATOR (101 UNIT)

Regulation	Measured ΔV		Worst Case ΔV	
	MV	Percent of 24 Volts	MV	Percent of 24 Volts
Load	27.7	0.115	72	0.30
Line	9.0	0.0275	12	0.05
Temperature	11.8	0.0492	48	0.20
Total	48.5	0.206	132	0.55

The data of Table 4-14 represents load changes from 0 to 72 ma, input buss changes from -26 to -36 volts, and temperature changes from + 50 to + 90°F. This temperature range corresponds to S2-0100 (18 February 63) Para. 3.4.1.1.3 Operative Within Specification. However, test data show satisfactory operation from -40 to +130°F, and Syncom I experience indicates that operation at +170°F would present no difficulty. The maximum measured temperature coefficient was ± 0.005 percent per °F average for -40° <T< + 130°F. This coefficient may be positive or negative, depending upon differential amplifier tolerances.

STATUS OF BILL OF MATERIALS

The preliminary Bill of Materials published on 29 April will be revised during the next period. Considerable revision will be made as a result of efforts to standardize semiconductor types. A standard semiconductor list has been published and is being checked for adequacy.

The total semiconductor types have been reduced from 65 to approximately 40 by comparing parts requirements with different design groups. Circuit drawings are being changed, and parts counts will be available for the next report.

TRAVELING-WAVE TUBE POWER AMPLIFIER

TWT Status

The major effort during this report period was expended solving manufacturing problems in preparation for the production run. The tube design has also been finalized.

A major manufacturing problem was in maintaining the desired tolerances on the helix pitch to increase yield. A uniform helix pitch is necessary to keep reflections along the helix structure at a minimum and thus maintain a short circuit stable tube. In addition, for the tubes to be reproducible, the pitch must be constant from tube to tube, because the beam voltage is related to phase velocity which in turn is a function of helix pitch. By a carefully controlled winding process and a careful selection of helix structures, a tolerance of ± 0.0002 inch can now be maintained to ensure good reproducibility and high yield in the production run. As a result of refining these techniques, only a small number of tube starts were made during this report period.

The final design of the 384H uses a helix pitch between that tested in tubes 384H-32 and 384-33. The optimum voltage of operation will be about 1260 volts. These tubes will be thoroughly tested next period when the production run will start.

Life Improvement Program

Present theoretical understanding of the oxide-coated cathode and empirical data obtained from the space tube program are sufficient to assure that the design of the 384 cathode will not fail due to normal depletion of either the cathode coating or the zirconium activating agent in the cathode base material. However, to obtain this designed life, it must be established that the cleaning and processing of the tube parts, cathode coating application, processing, and activation schedules are adequate to ensure that full use of the zirconium activating agent and barium oxide coating is possible. A diode life test program has been initiated to accomplish this.

Except for the helix assembly the diodes will have the same electron gun, envelope, and parts as the 384 traveling-wave tube. Therefore, the diode cathodes will have the same environment and operating conditions as the cathodes used in the actual tube. This will ensure that the data obtained from this program will be representative of the 384 cathode.

The specific areas that are being investigated are:

1) Several cathodes of each lot will be subjected to a complete composition analysis to determine the amount of carbon removal, any loss of activator material, and any pickup of contaminants which might have a deleterious effect on cathode life. These analyses will be taken after cleaning and washing, after wet hydrogen firing, and again after oxidation and reduction in dry hydrogen, which is the last step in the cathode cleaning procedure.

2) Platinum versus platinum — 10-percent rhodium thermocouples will be installed in each diode to determine the true cathode temperature within ± 5° C. This will allow the

determination of the effect of activation temperatures on the long-term emission capabilities of the cathode and provide a means of monitoring the cathode temperature during the entire life test program, thereby providing precise life failure data.

3) The eventual failure of the cathode will be accelerated in a controlled manner to obtain the life failure data in a reasonably short period of time. This will be accomplished by varying the cathode thickness, cathode temperature, and cathode loading in accordance with the following schedule:

   a) Three normal-thickness cathodes to be run at an elevated cathode temperature of 840° C and normal cathode current. Calculated theoretical life is 2 years.

   b) Five cathodes one-half normal thickness to be run at an an elevated cathode temperature of 840° C and normal cathode current. Calculated theoretical life is 1 year.

   c) Five cathodes one-half normal thickness to be run at an elevated cathode temperature of 840° C and three times normal cathode current.

   d) Three cathodes one-quarter normal thickness to be run at normal cathode temperature and normal cathode currents. Calculated theoretical life is 5 years.

   e) Four cathodes one-quarter normal thickness to be run at an elevated cathode temperature of 800° C and normal cathode current. Calculated theoretical life is 1 year

   f) Five cathodes one-quarter normal thickness to be run at an elevated cathode temperature of 840° C and normal cathode current. Calculated theoretical life is 3 months.

These first 25 diodes are under construction and are being made to the specifications outlined above. It is planned to construct 25 more diodes. They will be run under conditions where the preliminary data from the first 25 suggest that the sample size should be enlarged to provide significant data.

<u>TWT Power Supply (174 Unit)</u>

Parts procurement has delayed development of this unit. However, these parts are expected during the early portion of the next report period.

PHASED ARRAY TRANSMITTING ANTENNA

Sixteen antennas were assembled with the matching sections and adjustment screws described in the previous report. With the aid of the adjusting screws the input VSWR of most of the antennas could be set less than 1.5 from 3980 to 4200 mc. A few exceeded this near the edges of the band, but were still less than 2 to 1 VSWR. The latter apparently is due to the center conductor in the teflon dielectric slightly off-center position causing variations in assembly. A low-loss dielectric with a dielectric constant near that of teflon has been ordered for use as the spacers between matching sections and perhaps throughout the entire antenna.

The etched stripline boards for the two-level output couplers and the eight-way power divider are completed. One of the output coupler boards is shown in Figure 4-7 while the power divider is shown in Figure 4-8. All of the ground planes have been fabricated. Drill fixtures are being made so that all of the boards and ground planes can be drilled identically to ensure that proper registration will be obtained between the various units. Drawings for most of the miscellaneous hardware, such as supports, spacers, probes, etc., are nearing completion and fabrication of these parts has been initiated.

Details of the support structure to hold the phased array are being finalized. The primary support takes place at the ferrite phase shifters, which are the heaviest items. A method of supporting and stiffening the 16 antennas is also being finalized. A tentative technique is to fabricate a large fiberglass tube which will fit inside the 16-element array, to which the fiberglass sleeves on the individual elements will be fastened. A flat plate on top will support the receiving antenna.

PHASED ARRAY CONTROL ELECTRONICS

Status of PACE Circuitry

Seventy-five percent of the PACE final circuits were released in the month of June, bringing the total released to 95 percent. The VCO and integrator are the only two circuits for which final releases have not been made.

All product design of the Advanced Engineering Model (AEM) has been completed.

Forty-six of 62 cards have been fabricated and delivered to checkout; 16 cards are awaiting transistors for completion. Unit assembly for the PACE AEM is 60 percent complete (Figure 4-9).

Checkout of 28 percent of the AEM cards was completed, and checkout of 17 percent was partially completed by the end of June.

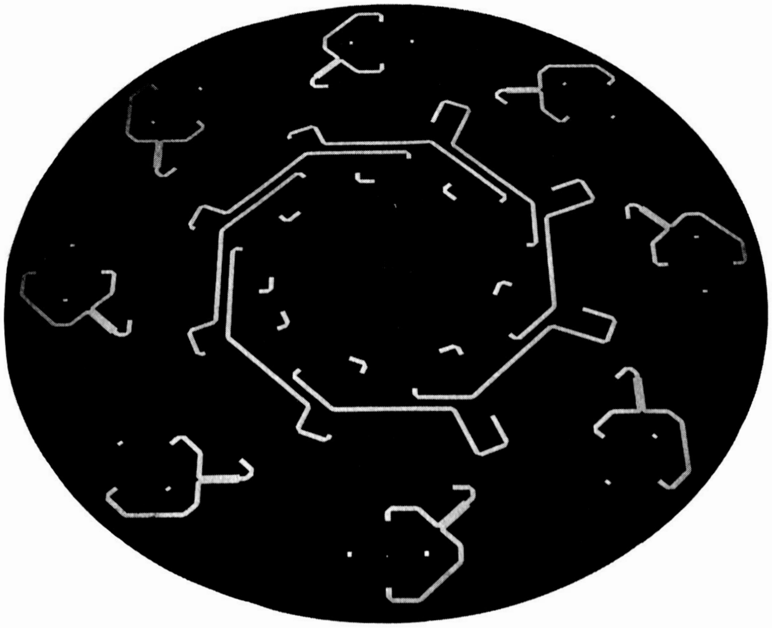


Figure 4-7. Output Coupler Stripline Circuit

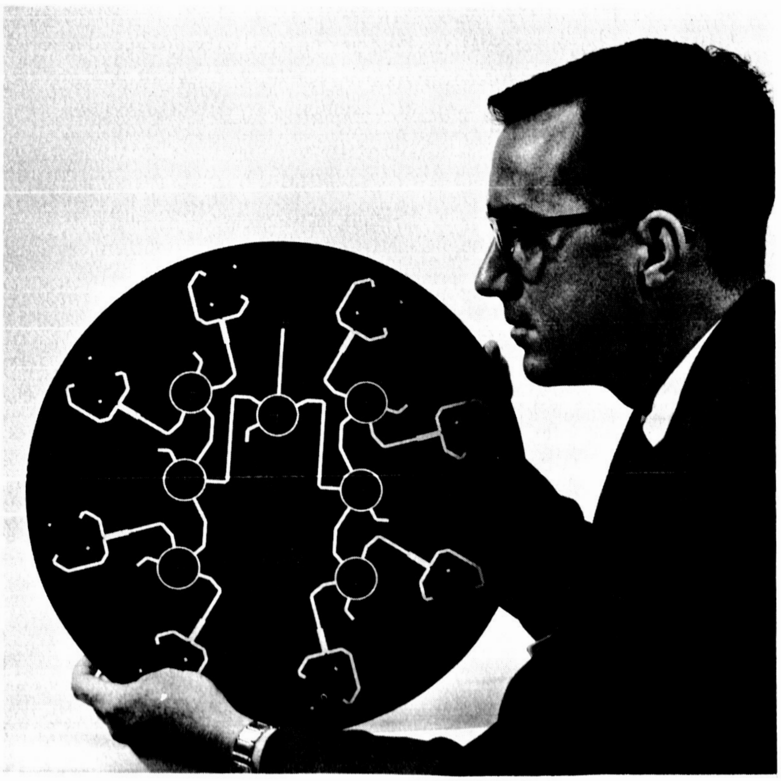


Figure 4-8. Eight-way Power Divider Stripline Circuit

Frequency Lock Loop

The frequency lock loop breadboard is 90 percent complete, incorporating final versions of those circuits that have been released. A stable psi-pulse generator has been designed and constructed for detailed test and evaluation of the frequency lock loop. A design procedure has been developed for synthesis of the frequency lock loop with unconditional stability.

Due to the nonlinearities in the frequency lock loop, the synthesis model does not readily provide the transient response of the loop. To aid in optimizing the loop parameters, a digital computer program has been written which simulates the frequency lock loop. The program will be used to evaluate different combinations of parameters prior to selecting the final values. The loop design, including final circuits, should be completed during the next reporting period.

Solenoid Driver and Pulse Jet Fire Angle Circuitry

Solenoid Driver

Circuit configuration was reviewed at Jet Control Electronics design review and approved. Four models of this driver will be fabricated and further testing of the Marquardt valves will be conducted.

Jet Control Electronics (JCE)

The review presented the fact that by sacrificing very little operational capability, the jet control could be accomplished with seven commands (instead of sixteen). This change would simplify the JCE logic and reduce the number of components by roughly 40 percent. The pros and cons of this suggestion are presently under evaluation.

Power Amplifier Regulator (163 Unit)

This regulator is in the process of being completely redesigned. Instead of ±24-volt and ±35-volt outputs, the voltage levels will be changed to approximately ±12 volts and ±20 volts. The new power supply will be unregulated. It will be connected to the unregulated bus through a resistor which will burn open similarly to a fuse in the event of a fault. The dc-dc converter will be capable of being started and stopped by command.

CENTRAL TIMING ELECTRONICS

Central Timer (Figure 4-10)

A preliminary design review for the central timer will be conducted early in the next report period. All circuit configurations are firm with the exception of the apogee motor driver. Among the problems as yet unresolved are:

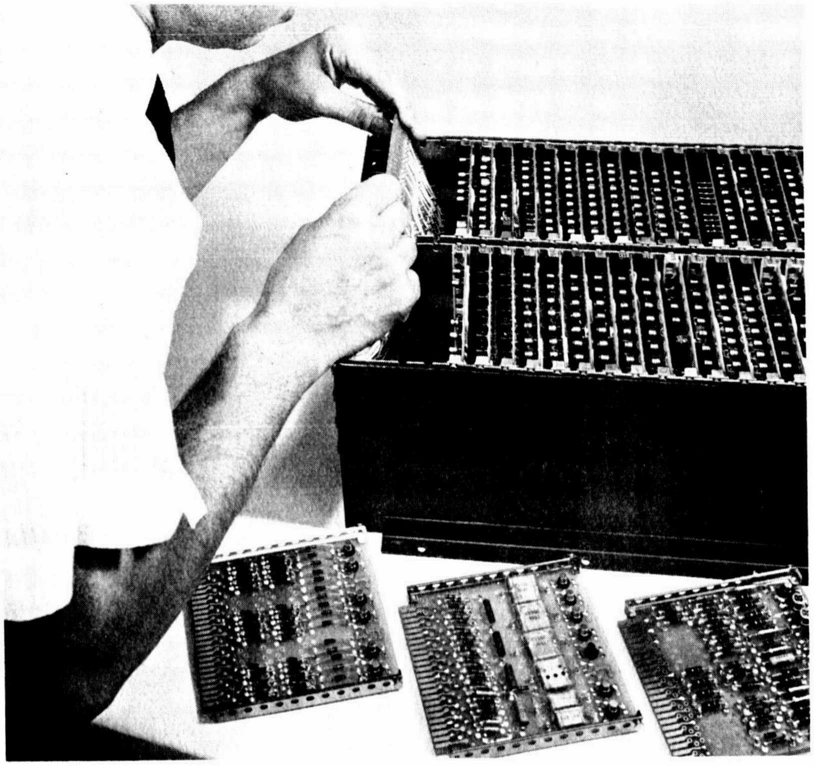


Figure 4-9. Pace Circuit Cards

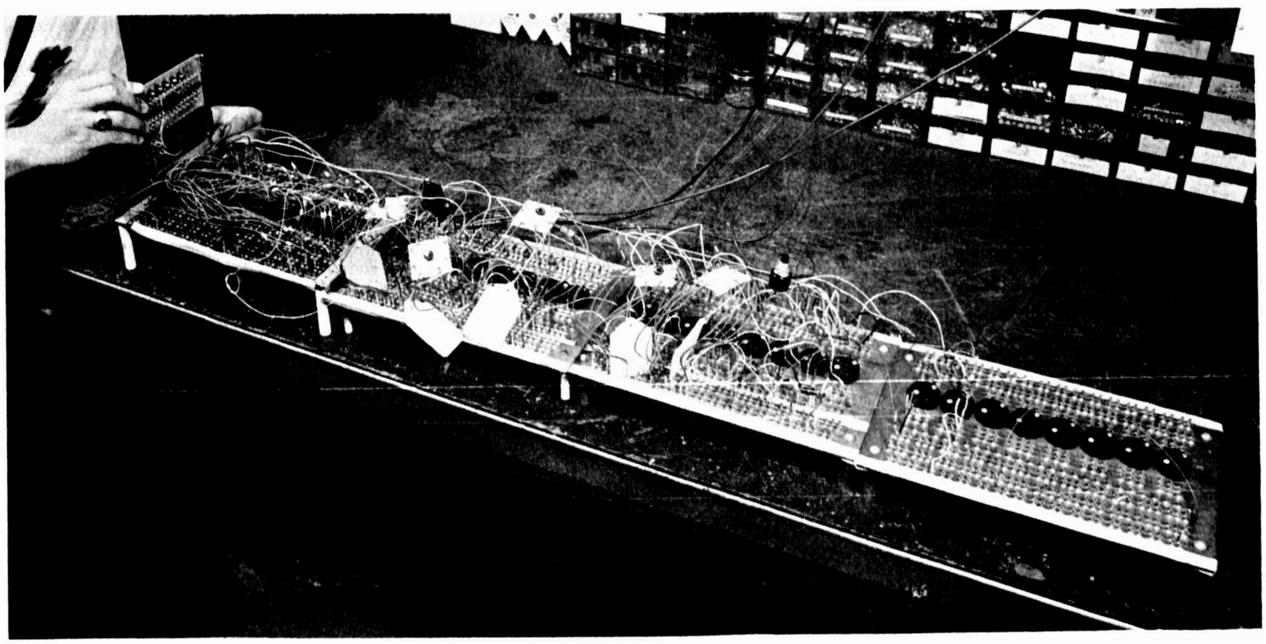


Figure 4-10. Central Timer Breadboard

1) The number of separation switches to be allotted per timer to assure enabling of each quadrant's central timer at separation;

2) the method of enabling the apogee motor driver; and

3) accuracy required for the 315-minute apogee timer output (this affects the core reset configuration).

Fabrication of the breadboard of the central timer is approximately 90 percent complete. Full completion awaits delivery of the required transistors.

Design of an all-electronic apogee motor driver is being considered, in addition to the driver configuration utilizing pyrotechnic switches in series with the squib-firing output. It is felt that a lighter, more reliable driver may result from such a design.

Tests are now being initiated to determine the long-term stability of intermediate counts within a core. The plan is to incrementally increase the time interval between full-scale counts and note whether there is a maximum time interval beyond which unreliable operation results.

Means and Control of Firing Apogee Motor

Previous planning for theapogee motor firing was to have the firing by timer only. Any two of the four timers could fire the engine. This approach however, is inflexible and NASA requested consideration of a command backup to the timers. Part of the concern was caused by the fact that it was unknown whether there would be two, three, or four quadrants. Whether or not there is a command backup, it is recommended that four timers be used. The timers will represent token weight, volume, and power requirements, and their incorporation should be very simple.

There is no particular problem in firing the apogee engine by command. The cost is primarily in harness wires, one wire per decoder being added. If such a capability is built into the system, it is recommended that a two-command sequence (the commands being received with a relatively short time separation) be used. This would be implemented rather simply in the decoders and in the ground equipment.

COLLINEAR ARRAY (CLOVERLEAF) RECEIVING ANTENNA

The 6165-mc model of the cloverleaf array has been fabricated. Initial testing has been completed, which included measuring the input impedance, antenna gain, and E-plane and H-plane antenna patterns. The results of these measurements are presented in Table 4-15.

TABLE 4-15. MEASUREMENTS FOR 6165-MC CLOVERLEAF ARRAY

Frequency, megacycles	VSWR	Omnidirectionality, decibel	Average Beam Width, degrees	Highest Sidelobe Level, decibel	Gain, decibel
6015	1.09	0.6	18.9	12.8	7.6
6065	1.38	0.5	18.6	12.3	7.5
6115	2.20	0.6	18.8	12.1	7.2
6165	2.30	0.4	18.4	11.9	6.6
6215	2.10	0.6	18.1	11.1	6.1
6265	2.25	0.3	18.2	11.3	5.6
6315	2.30	0.5	18.2	12.1	5.6

The array is resonant at a frequency of about 6015 mc; therefore, some means will be devised to match the array to improve its gain at the center frequency.

VELOCITY AND ORIENTATION CONTROL

Analysis of Spacecraft Orientation Maneuver

A revised error analysis of the Syncom II orientation and velocity increment maneuvers is in progress. The alignment and parameter uncertainties of the ψ and ψ_2 sun sensors are being considered as basic error sources in the determination of vehicle angular orientation. For example, a misalignment, ΔI, between the planes of the ψ and ψ_2 sun sensors results in an error in the measurement of ϕ, the angle between the spin axis and LOS line of sight to the sun, given by

$$\Delta\phi = \left[\sin \psi_2 \csc^2 I \sin^2 \phi\right] \quad \psi I$$

The random position error of a commanded thrust pulse will be developed as a function of engine and electronic errors. The work of Technical Memorandum 649, by D. D. Williams, will be extended so that the sensitivity of Rhumb-line attitude motion and vehicle velocity increment to initial apogee conditions, observed orientation, thrust pulse position, and vehicle dynamic unbalance will be obtained analytically as a first perturbation. Approach will be checked by the IBM 7090 attitude trajectory program discussed in the previous (May) report.

In this manner the effect of component and subsystem errors in gross vehicle motions may be directly observed, and conversely, the performance specifications (tolerances) placed on those items may be re-evaluated in terms of system performance.

Bipropellant Rocket Reaction Control System Development Status

A major portion of the effort at Marquardt was spent in preparation for delivery of the engineering model. Performance specifications have been completed on the relief valve, pin puller, propellant filter, calibrating orifice, injector solenoid valves, fill and drain nipple, pressure transducer, swivel mechanism engine, tank, and the engineering model. Acceptance tests have been completed on all components except the engines, tanks, and swivel mechanism.

The engine acceptance test program has included pulse mode runs of 3 minutes and 6 minutes duration for the one-third duty cycle (1 second ON and 2 seconds OFF) and the one-sixth duty cycle (0. 1 second ON and 0. 5 second OFF), respectively. Tests were conducted at both the 3- and 5-pound thrust levels. A preliminary analysis of the test data indicated three discrepancies:

1) A thrust transient period was noted during the initial 25 to 35 pulses of a series for the one-sixth duty cycle. During this period, thrust gradually increased to a steady level

approximately 20 percent greater than the starting level. The increase in thrust is attributed to oxidize leakage around the outside of the calibrating orifice.

2) One solenoid valve failed to open during a one-sixth duty cycle run, but functioned properly during subsequent runs. A failure investigation has been initiated.

3) In the test which preceded the solenoid failure, the thrust level gradually decreased to approximately 50 percent of the established 5-pound level, then gradually increased to the original thrust. It is believed that this occurrence is associated with the subsequent valve failure. The engine acceptance tests will be rerun during the first portion of the next report period.

The engineering model was assembled exclusive of the engines and swivel mechanism. A nitrogen pressure check disclosed leaks in two of the four tanks. The tanks did not leak during hydrostatic test prior to assembly. One tank leaked in the vicinity of a weld and has been repaired. The second tank leaked through the flange of a fitting. Metallurgical analysis showed that the fitting had been machined such that the elongated 17-7 PH grain structure was oriented transversely to the flange resulting in material porosity. The fitting has been repaired by welding. In the future fittings will be fabricated with the grain oriented longitudinally with the flange.

The Marquardt Company is scheduled to complete Phase I of the development on 15 July.

Marquardt Report (Hughes-Edited)*

Engineering

Design. Design support activity included engineering drawing release and maintenance, development test support, and manufacturing support.

Drawing release, changes, and maintenance were accomplished on the following:

1) Engineering model assembly: Drawing was revised to include minor changes as a result of the actual assembly buildup. An adjustable restrictor design employing a pinch clamp and small seamless tube was detailed and released for fabrication in place of the previous variable restrictor design. Minor changes were made to the radiation shield to minimize fabrication problems.

2) Rocket engine assembly: Changes made on the injector head drawing included design improvements to eliminate oxidizer

*The information in this subsection was abstracted from Marquardt Monthly Progress Report MR-1-4, dated 15 June 1963.

boiling and injector insert blowout. The oxidizer boiling problem has been minimized by incorporating a thin wall stainless steel hypodermic tube brazed at both ends and insulated from the head by a space gap. The tube inside diameter is a constant 0.019. The fuel injector insert was replaced by a drilled hole with a conical inlet; hole size remained 0.016 diameter. In addition, the "k" seal contact area on the head sealing surface was reduced by 30 percent to minimize heat conduction from the combustion chamber. The oxidizer injector valve-to-head contact area was reduced also to minimize heat conduction to the valve from the head. High-pressure drop-fixed restrictors were added to both the fuel and oxidizer legs of the engine, located at the downstream end of the injector valves.

3) Flight-type tank assembly: Drawing was changed to improve pressure testing technique performed during heat-treating cycle. Post-heat-treat cleaning requirements were also revised to control time in chemical solution.

4) Fixed restrictors: A drawing was released for a family of high-pressure drop-fixed restrictors for use in the engine assembly. Placement of these restrictors in the engine eliminated the need for them in the system plumbing.

In development test support, additional design support was required during this period for various engine tests. Engineering orders, EEOs, and TWEOs were issued in support of test evaluation of various design features incorporated in the injector head. The flight-type tank development structural test plan is 80 percent complete and will be completed early in the next reporting period. The long-term storage test plan will also be completed in the next reporting period.

Engineering effort was expended in support of manufacturing activities on the following: tanks (flight type), injector heads, development engines, components (outside manufacturer), and engineering model assembly (buildup).

Development. Engine tests were conducted with an injector head modified to obtain higher injection pressures. This was accomplished by decreasing the injection orifice size. Test results with this head using both MON-15 and N_2O_4 as the oxidizer (fuel, monomethyl-hydrazine) showed that the total impulse per pulse ($\int F\,dt$) decreased during continuous pulse mode operation at either the 1 second on, 2 second off, or 0.1 second on, 0.5 second off, duty cycles. The decrease in total impulse was accompanied by a decrease in oxidizer-to-fuel ratio indicating that the oxidizer vapor pressure was exceeded in the injector head as the engine temperatures increased.

An injector head change to reduce heat transfer to the oxidizer flowing through the head was made by reducing contact area between the head and the oxidizer passage. This modification showed a considerable improvement in the impulse per pulse characteristic for continuous pulse mode operation.

A change in the head manufacturing and injection techniques was incorporated to improve the injector orifice alignment and provide more consistent spray pattern characteristics. The fuel injector orifice insert concept was replaced by machining the orifice directly into the injector head to eliminate any possibility of losing the injector during engine firing.

Engine tests with the latest injector head configuration have indicated changes in the dynamic characteristics of the fuel manifold pressure (i. e., pressure directly upstream of the engine) during continuous pulse mode operation. The transient response of the fuel pressure indicates the possibility of gas entrapment between the fuel solenoid valve and restrictors (required for system calibration) upstream of the solenoid valve. The restriction in the line is being moved downstream of the solenoid valve to improve the system dynamics. This change also provides higher propellant pressures in the solenoid valves during engine operation and decreases the possibility of propellant vaporization in the solenoid valves.

Tests on the breadboard model were successfully completed during the report period. The breadboard feasibility test witnessed by personnel from NASA and Hughes was conducted in the TMC Cell 9 facility. Each engine, radial and axial, was fired at the appropriate duty cycle to obtain a minimum of 2-1/2 minutes of actual burn time. During engine firing, the complete breadboard model was spinning at rates varying from 75 to 125 rpm. At the conclusion of the breadboard feasibility tests the axial engine was removed from the breadboard, installed in the engine test cell, and fired to verify engine performance levels. A visual inspection and leakage test of the axial engine after the performance test showed small cracks in the combustion chamber. The expansion cone of the radial engine combustion chamber was broken after completion of the breadboard feasibility tests and could not be tested at altitude conditions. The investigation of both combustion chambers is continuing to determine how the damage occurred. Two possibilities appear feasible: an unstable shock system occurred in the diffusion section of the nozzle due to operation of the altitude nozzle at sea level conditions; or the engines were improperly handled during the installation and test phase.

All components for the engineering model, excluding engines, have been acceptance tested and mounted on the Hughes structure. Acceptance tests on the engines and the pressure drop calibration of the subsystem are expected to begin during the week of 17 June 1963.

The objectives and results of the engine tests are described in detail below and summarized in Table 4-16.

1) Engine Test 15: To determine engine temperatures and performance for the radial engine configuration (i. e., with heat shield using 0.013-inch-diameter injector orifices).

The test was conducted at altitude conditions using MON-15 and monomethyl-hydrazine as the propellants. Engine performance at a 4.3-pound thrust level was determined for 1- and 0.1-second runs (I_{sp} = 261 and 232 seconds respectively) and an 11-minute run at a duty cycle of 0.1 second on, 0.5 second off, was made to determine steady-state temperatures. The data plotted in Figure 4-11 show the decrease in impulse per pulse during the run. A 16-minute run at the above duty cycle and a lower thrust level (~ 3 pounds) indicated the same trend. An analysis of the data indicated little or no improvement in the decreasing impulse with time problem (probably boiling) as a result of decreasing injector orifice diameters.

2) Engine Test 16: To determine the performance of the radial engine configuration (with heat shield) using nitrogen tetroxide and monomethyl-hydrazine as the propellants. N_2O_4 was the oxidizer used in all engine tests following test 16.

The engine configuration for this test was identical to that of Test 15 using the MON-15 and monomethyl-hydrazine. Performance for 1-second and 0.1-second runs was 268 and 220 seconds respectively. The impulse per pulse for long-duration runs at the 0.1 second on, 0.5 second off, duty cycle showed a trend similar to that in Test 15. Since the smaller injector orifice sizes did not eliminate the oxidizer boiling problem with either MON-15 or N_2O_4, a modification to the injector head was made to decrease heat transfer to the oxidizer.

3) Engine Test 17: To determine if an injector head modification decreased the tendency for oxidizer to boil in the injector head during pulsing; and to determine if variables in the altitude cell operation influenced engine temperatures.

The oxidizer side of the injector head was modified by replacing the oxidizer insert with a tube extending from the oxidizer solenoid to the combustion chamber face. This tube brazed to the injector head at each end was insulated from the head by clearance space throughout the remaining length. A pulse test at the 0.1 second on, 0.5 second off, duty cycle was made for a duration of 7.8 minutes. The thrust information from this test was not valid due to a problem in the thrust instrumentation. An analysis of the combustion chamber pressure

TABLE 4-16. ENGINE TEST SUMMARY

Test Number	Test Date (1963)	Injector Head	Combustion Chamber	Test Conditions	Number of Runs	Total Burn Time, Seconds	Maximum Single Run Time	Remarks
15	5/13 5/14	X19151 S/N 003	X19158 S/N 004	Altitude	26	191	16 minutes*	Test objective completed
16	5/15	X19151 S/N 003	X19158 S/N 004	Altitude	21	479	16.6 minutes*	Test objective completed
17	5/20	X19151 S/N 003	X19158 S/N 004	Altitude	31	237	8.7 minutes*	Test objective completed
18	5/22	X19151 S/N 006	X19158 S/N 004	Altitude	15	19	2 seconds	Test objective completed
19	5/25	X19151 S/N 002	X19158 S/N 007	Altitude	34	217	5.8 minutes* 3.0 minutes**	Test objective completed
20	5/28	X19151 S/N 001	X19158 S/N 004	Altitude	14	14	1 second	Test terminated due to loss of the fuel insert
21	6/4	X19151 S/N 005	X19158 S/N 004	Altitude	31	526	17.5 minutes* 9 minutes**	Test objective completed
22	6/5	X19151 S/N 005	X19158 S/N 004	Altitude	24	193	15.2 minutes*	Test objective completed
23	6/7	X19151 S/N 005	-----	Altitude	13	331	290 seconds	Test terminated due to decay in engine performance

* Continuous run at duty cycle 0.1 second on, 0.5 second off.
** Continuous run at duty cycle 1 second on, 2 seconds off.

indicated that thrust did not decrease as much as with the unmodified injector head although head temperatures were approximately the same.

Runs were made during the test to determine the effect on engine temperature of small amounts of nitrogen flow into the altitude chamber and to determine if exhaust gas recirculation contributed to engine temperatures. The test data indicated that there was no significant effect on engine temperatures due to the nitrogen flow or to recirculation.

During the final pulse run, a change in pulse characteristic was observed. Subsequent visual examination of the engine showed the fuel injector insert was out, and examination of the fuel manifold pressure oscillograph trace for the final run confirmed that the insert had come out during this run.

4) <u>Engine Test 18</u>: To determine engine performance as a function of pulse width to establish acceptance test criteria for a subsequent engine to be used on the breadboard feasibility test.

The engine was assembled with an injector head modified by replacing the oxidizer injector with a tube similar to the injector configuration of Test 17. Tests were conducted for pulse widths from 0.1 to 2 seconds; the reduced data are shown in Figures 4-12 through 4-15.

5) <u>Engine Test 19</u>: To determine axial engine performance following the breadboard performance demonstration.

The engine and swivel mount assembly were removed from the breadboard model and installed in the ATL test facility with the swivel mount locking pin engaged. Engine firing tests were conducted for pulse durations of 0.1 to 2 seconds; results are shown in Figure 4-16.

6) <u>Engine Test 20</u>: To determine engine performance of injector head reworked to obtain close tolerance alignment of the injector orifice.

A rework of the injector head was made to eliminate spray pattern inconsistencies previously observed and to improve the alignment and impingement point. Physical inspection and observation of the spray pattern verified that injector alignment had been improved.

The engine was installed in the test cell and the initial runs to establish run condition indicated performance was low. The test was terminated and a visual inspection of the engine showed

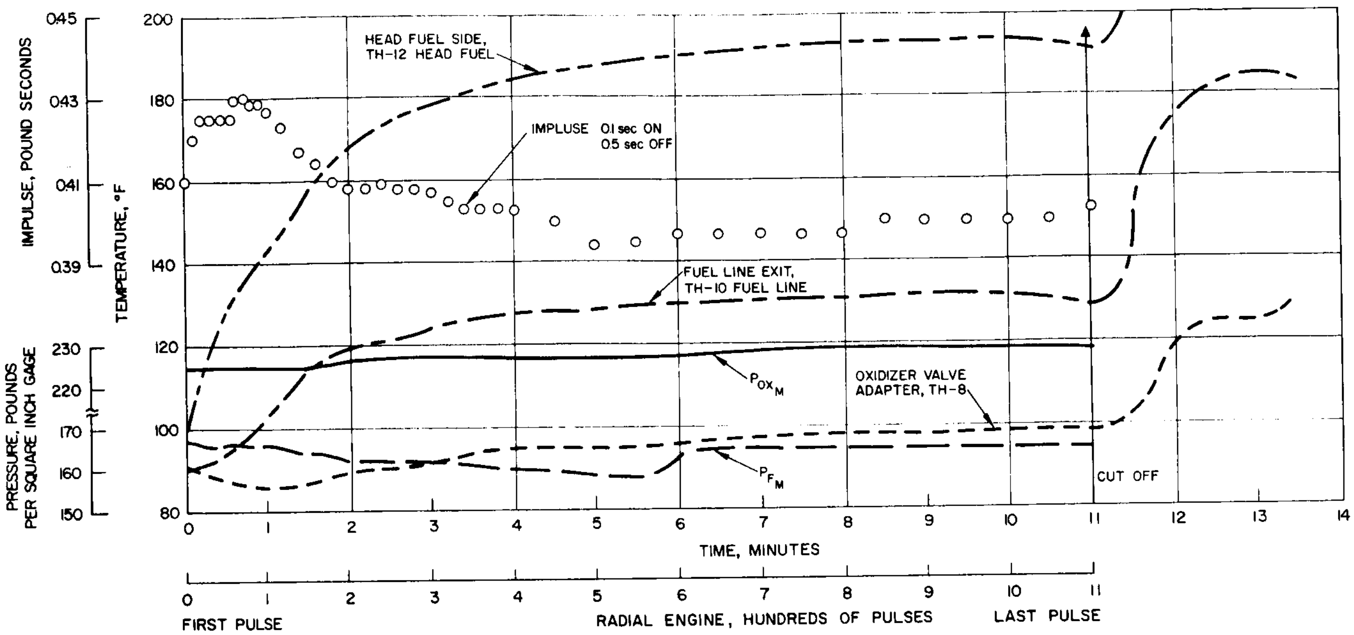


Figure 4-11. Radial Engine Test Run 8, 13 May 1963 (1/6 duty cycle)

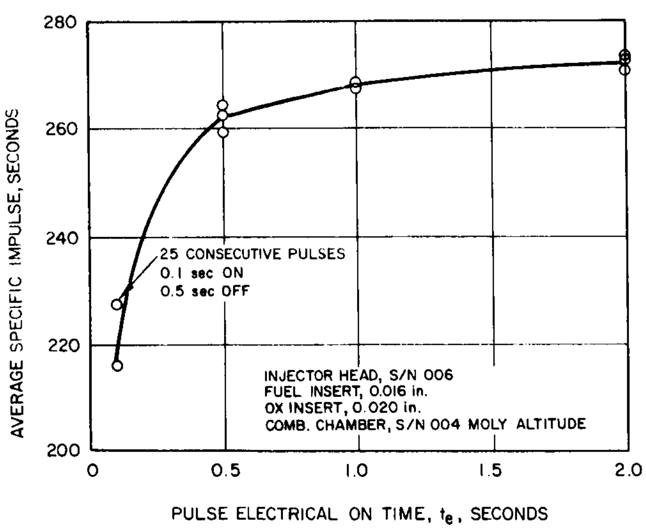


Figure 4-12. Syncom 5-Pound Engine Performance Test

Test 18, 22 May 1963

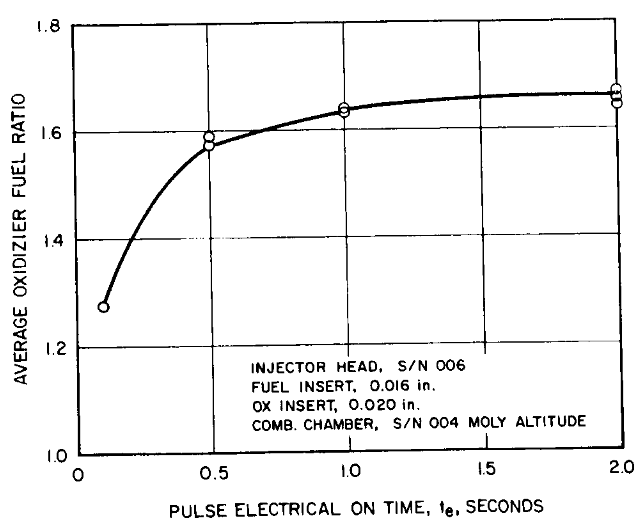


Figure 4-13. Syncom 5-Pound Engine Test, Mixture Ratio

Test 18, 22 May 1963

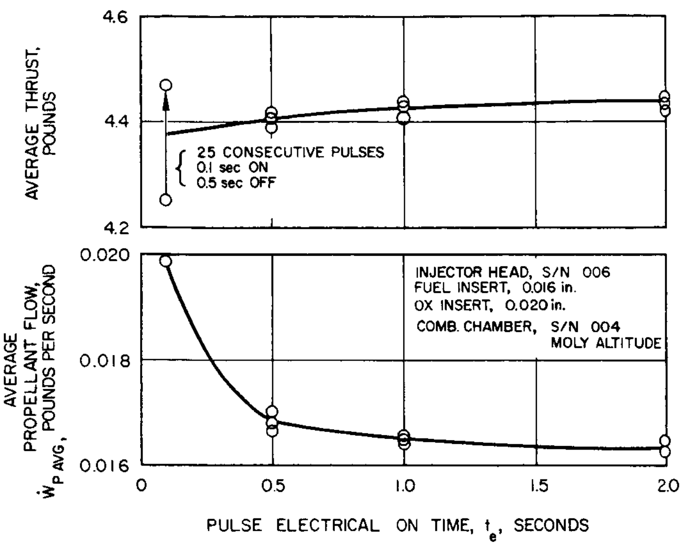


Figure 4-14. Syncom 5-Pound Engine Test, Propellant Flow and Thrust

Test 18, 22 May 1963

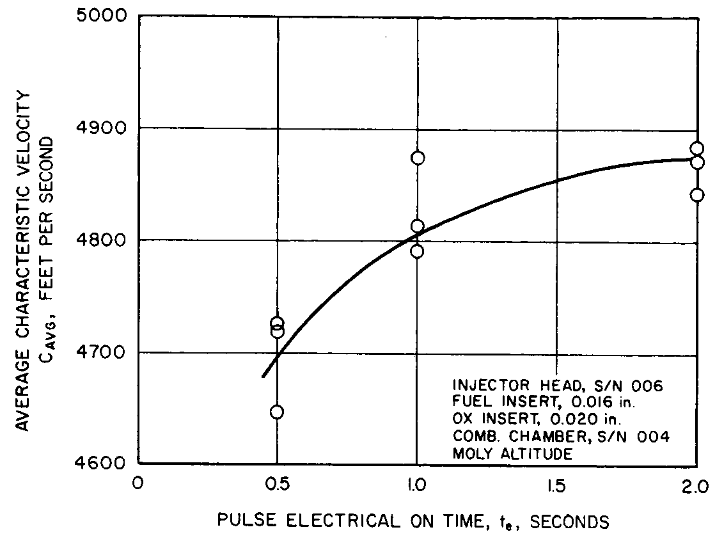


Figure 4-15. Syncom 5-Pound Engine Test, Characteristic Velocity

Test 18, 22 May 1963

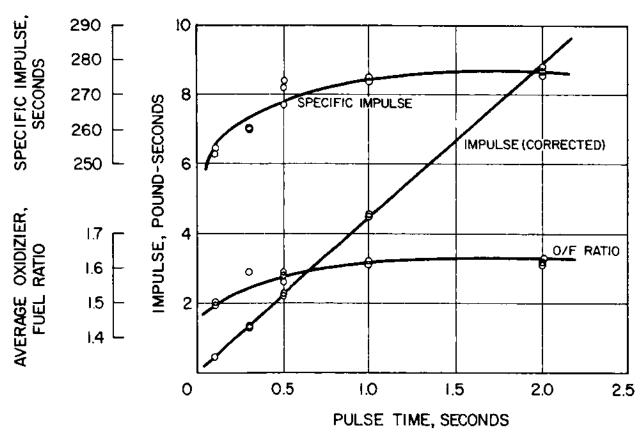


Figure 4-16. Syncom 5-Pound Engine Test

Test 19, 25 May 1963, Runs 6 through 20

that the fuel orifice had come out of the injector head, even though special care in fitting and staking the insert had been used in the manufacturing process. A review of the test data indicated that the fuel orifice had come out during the propellant cold flow operation conducted prior to the engine runs. Action was taken to eliminate the insert concept for the fuel orifice to prevent recurrence of the failure.

7) <u>Engine Test 21</u>: To evaluate injector head with fuel orifice as an integral part of the injector head and determine engine performance with close tolerance orifices; and to obtain combustion chamber temperatures.

The head modification consisted of drilling the fuel orifice in the injector face rather than using an insert. This change coupled with the oxidizer tube modification prevents loss of either orifice.

One-second runs made to determine engine performance indicated an I_{sp} of 270 seconds. Runs of 5 seconds duration were made to obtain the nozzle temperature transients as comparison for subsequent tests to be made with this injector head and a tantalum combustion chamber. The nozzle temperatures measured during the test were in agreement with nozzle temperatures obtained from previous tests.

The engine parameters for a continuous pulse run at a duty cycle of 0.1 second on, 0.5 second off, are plotted in Figure 4-17. The change in the impulse characteristic during the first 6 minutes of the run time is attributed to a change in the characteristics of the fuel manifold pressure response. The thrust, chamber pressure, and manifold pressures of Figure 4-17 are values taken at the end of the run and are not representative of the changes observed in the impulse which is thrust integrated over the duration of the run. The change in the characteristic of the fuel manifold pressure is believed to have resulted from gas entrapment between the engine and a facility valve that is used to adjust the pressure drop between the tank and the engine. It was concluded that additional information was required necessitating a rerun of this test.

8) <u>Engine Test 22</u>: To continue investigation of the dynamic characteristics of the fuel manifold pressure. This was a repeat run of the previous test using the same hardware.

Tests were made with and without the facility restricting valves in the facility system to determine the fuel response characteristic, which was also investigated at various manifold pressure levels. As a result of the testing the location of pressure drop elements was changed from upstream of the engine to a location downstream of the injector solenoid valves to maintain high pressures in the solenoid valves during runs. This modification will decrease the possibility of vaporizing oxidizer in the solenoid valve due to temperature increase and is expected to provide improved dynamic response characteristics.

9) Engine Test 23: To determine engine steady-state temperature for runs of continuous duration using a Sylcor coated tantalum combustion chamber.

The combustion chamber used in the test was identical in configuration to the S19158 chamber used in previous engine tests. Thermocouples were installed on the chamber similar to Test 21. Pressure drop elements were left upstream of the engine to provide direct comparison with Test 21 results.

Following a series of 1-second runs to establish run conditions (O/F = 1.62, thrust = 4.5 pounds) a continuous run of 30 seconds was made to determine the nozzle temperatures. The maximum indicated temperature during the run was approximately 4200°F. A second run at a thrust level of approximately 3 pounds thrust was initiated and continued for 4.8 minutes. The maximum temperature during the 3-pound thrust run was 3020°F. Chamber deterioration was evinced during the run by a decay in both the thrust and combustion chamber pressure, and visual inspection of the engine after the test showed a longitudinal burnout in the chamber. Considerable nozzle material had passed through the throat.

The breadboard model depicted schematically in Figure 4-18 was installed in the Cell 9 facility at TMC for the feasibility demonstration. Figure 4-19 shows the breadboard model mounted on the spin table in the test chamber. Propellants (MMH and N_2O_4) were loaded into the breadboard model tanks and preliminary firing tests were made to check engine operation and facility instrumentation.

The formal demonstration, witnessed by personnel from NASA and Hughes, was conducted 24 May 1963. During the demonstration the axial engine was fired at a duty cycle of 1 second on, 2 seconds off, for a period of 7-1/2 minutes. The actual engine on-time was 2-1/2 minutes. During the firing of the axial engine the table spin speed was varied between the limits of 75 and 125 rpm to demonstrate operation of the engine through the full range of deflection of the swivel mount. A time history of the spin speed, angular position of the axial engine, and combustion chamber pressure is shown in Figure 4-20.

The radial engine was fired at a duty cycle of 0.1 second on, 0.5 second off, for a period of 15 minutes to obtain 2-1/2 minutes of engine

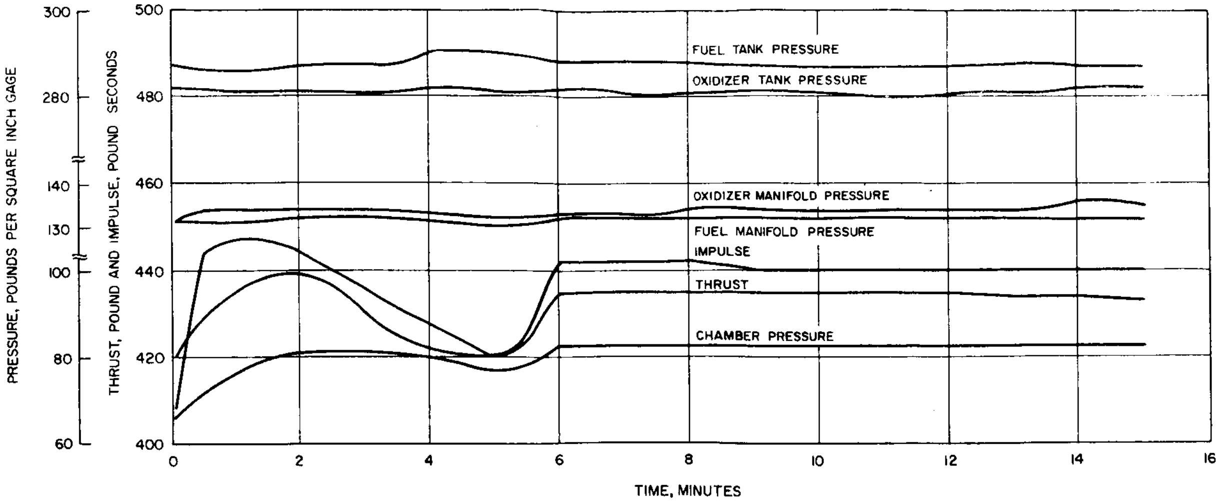


Figure 4-17. Syncom 5-Pound Engine Test
Test 21, 4 June 1963 (1/6 duty cycle)

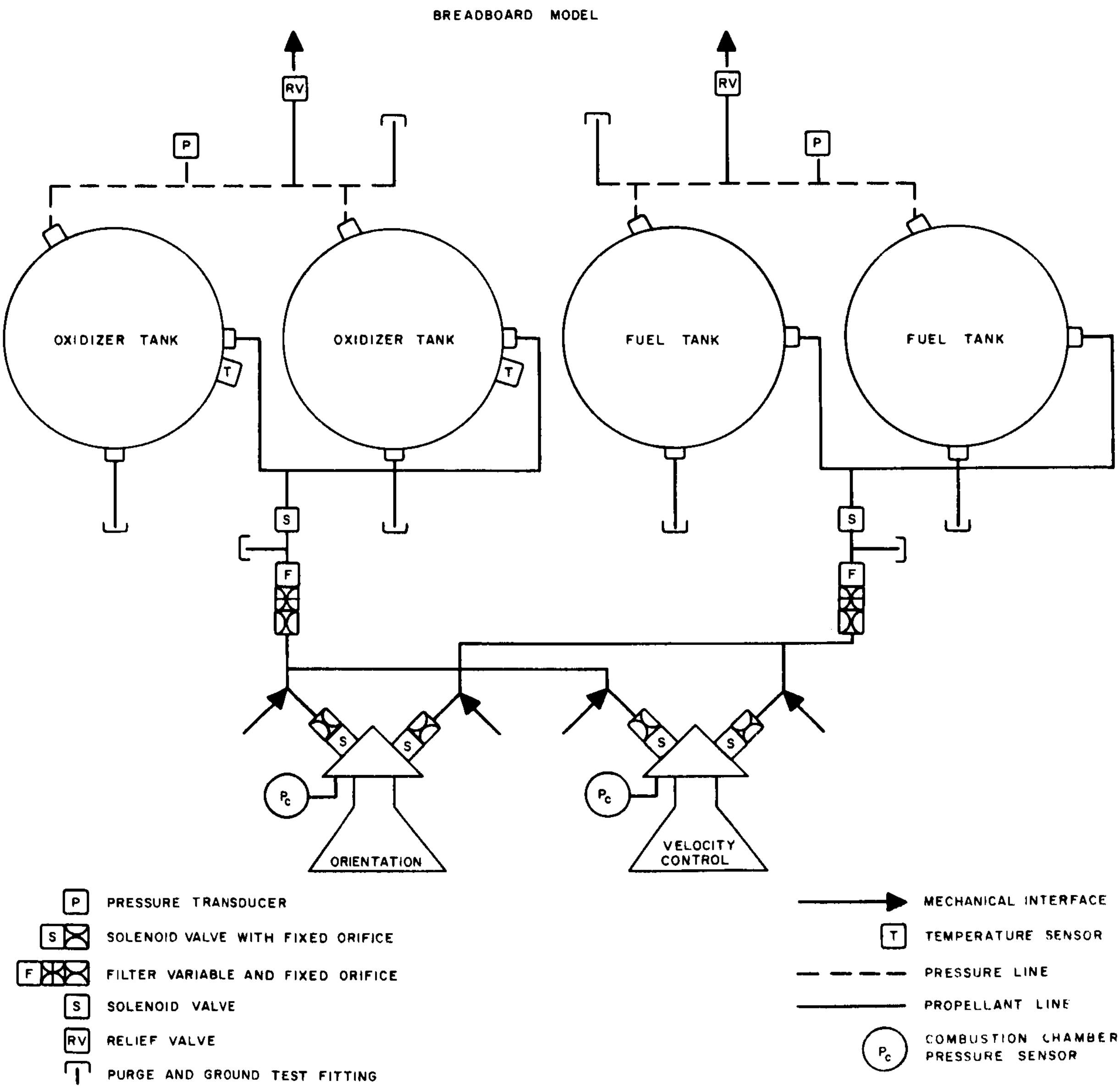


Figure 4-18. Breadboard Model Schematic

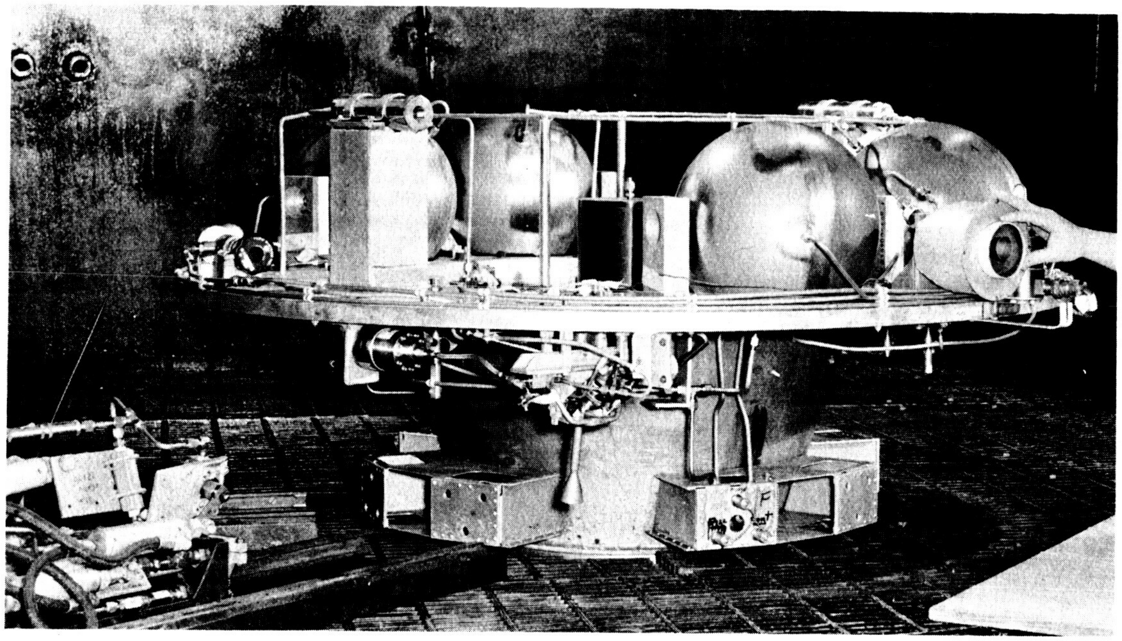


Figure 4-19. Breadboard Systems

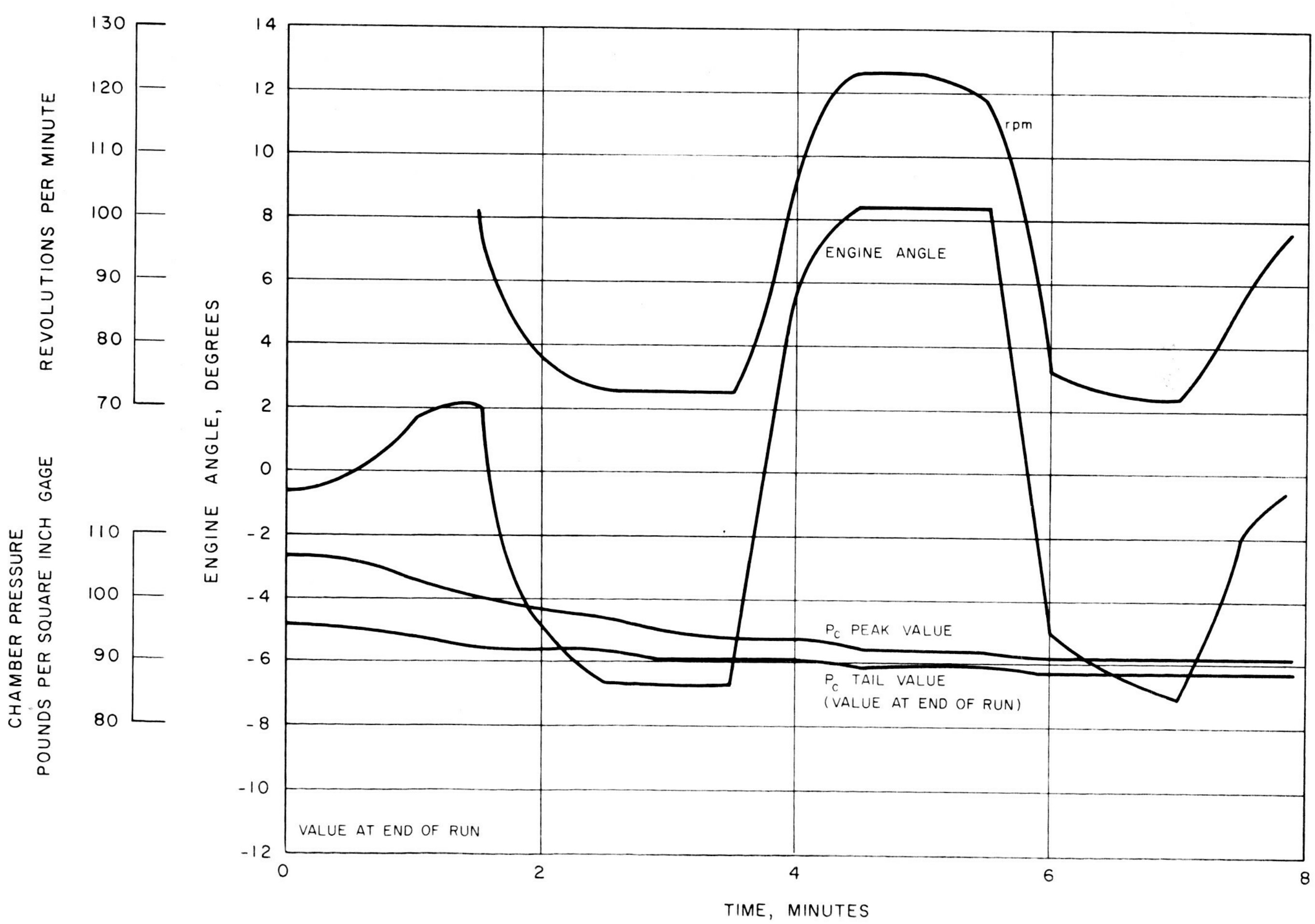


Figure 4-20. Axial Engine Breadboard Test
24 May 1963 (1/3 duty cycle)

on-time. The spin speed of the breadboard model during the radial engine firing tests was 100 rpm.

The location of the combustion pressure transducer for the radial engine, coupled with the short-duration pulse, did not provide a steady-state value. However, the consistency of the radial engine performance could be observed from the transient of the chamber pressure.

At the conclusion of the feasibility demonstration spin tests the axial engine was removed from the breadboard complete with swivel mount assembly and tested in the ATL engine test cell. The results of this test are described under Engine Test 19 and the performance curves are shown in Figure 4-16.

A visual inspection of the engine following the engine test showed fine cracks in the expansion section of the nozzle. A subsequent pressure check of the combustion chamber showed leakage along a fine crack in the combustion chamber portion of the nozzle.

The flexure-type pivot bearings of the axial engine swivel mount were damaged at the conclusion of the breadboard testing. An investigation of the cause of damage showed that the method of mounting the engine/swivel mount assembly on the breadboard (i. e. , with the engine vertically down) resulted in compressive loads on the pivot flexures rather than tension loads. The orientation of the flexure pivots in the swivel mount was designed for the engineering model in which the engine is pointed vertically upward with resultant tension loads on the pivot flexures. It is believed that the hysteresis of engine angle versus spin table rpm shown in Figure 4-20 resulted from the damaged flexures. No problems are anticipated for the swivel mount flexure pivots on the engineering model.

At the conclusion of the breadboard testing the breadboard assembly was moved from Cell 9 to a storage area. Visual inspection showed the nozzle of the radial engine was broken as shown in Figure 4-21. Although the physical evidence indicated that the nozzle may have been broken due to handling, examination of the break showed that the nozzle could have been cracked during the engine testing.

The investigation of the cracks in the axial engine combustion chamber and the break in the radial engine chamber is continuing. The location of the break on the radial engine indicates a possible cause may have been an unstable shock in the nozzle expansion section caused by firing the altitude-configuration nozzle at sea level conditions.

Analysis. Test results with the original oxidizer injector design (a 0. 033-inch diameter duct 1 inch long drilled in the steel head downstream of the oxidizer valve seat which feeds a 0. 020-inch-diameter orifice insert) resulted in boiling of the oxidizer in the injector head during operation at both the 3-pound thrust level and the 5-pound thrust level during pulsing

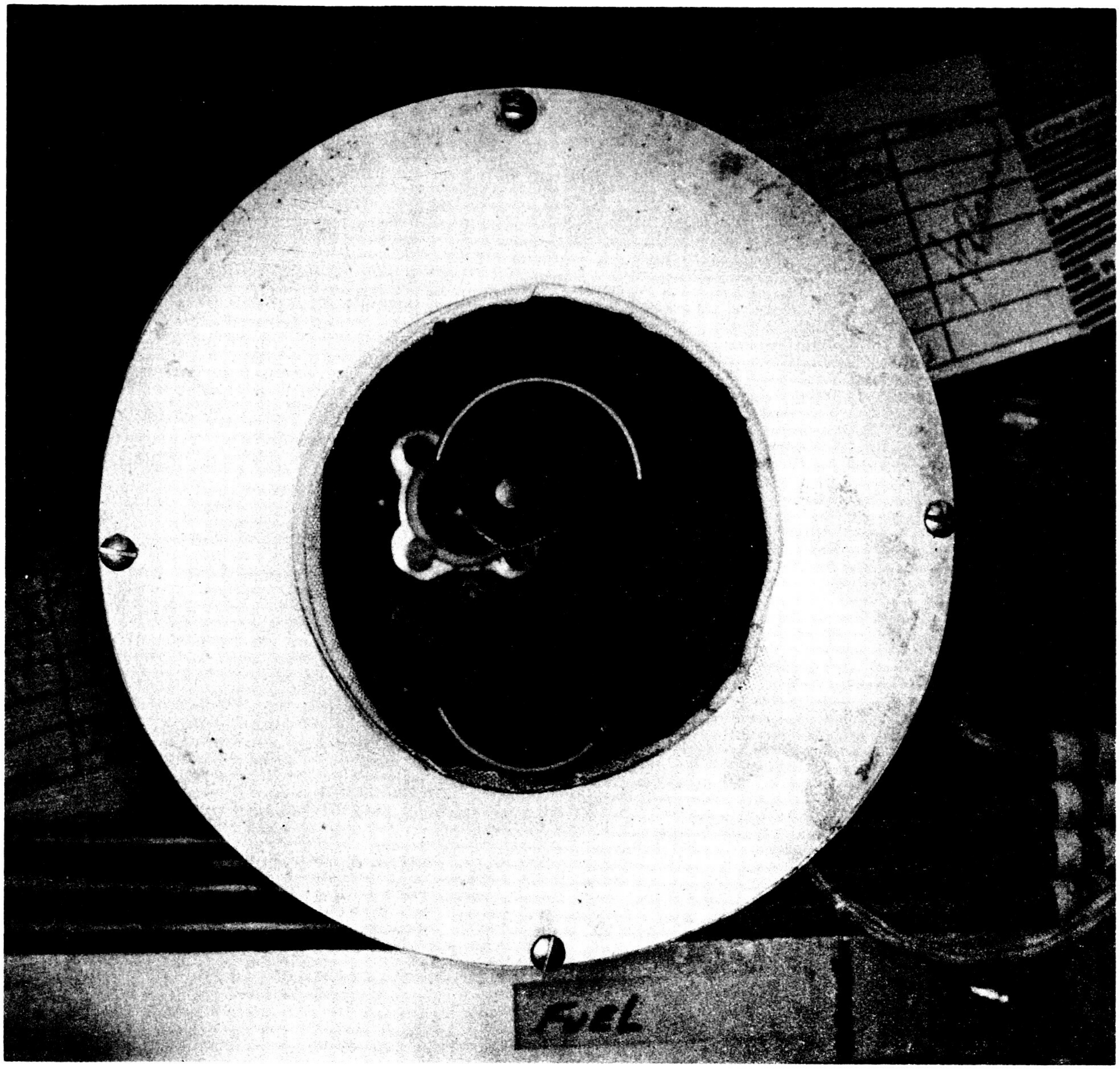


Figure 4-21. Breadboard Radial Engine and Heat Shield

operation at a one-sixth duty cycle. Analysis indicated that the primary variable in inducing boiling on the N_2O_4 in the injector head is the wall temperature in contact with the oxidizer. The effect of pressure level and velocity of the oxidizer in the duct is secondary. The boiling of the oxidizer in the injector head has been eliminated or minimized during operation by incorporating a low-heat-capacity tube (minimum wall thickness) between the valve and the injection point which is insulated from the head by a space gap. The flowing oxidizer can reduce the tube temperature to less than that which would induce boiling within 0.030 second at the 3-pound thrust level for a pulse at maximum soakback temperature, if the trapped oxidizer in the valve is below 125°F.

A tradeoff study based on available data was conducted to improve axial engine life. By changing to a pulsing duty cycle, a molybdenum engine life expectancy of 2 hours at a 60 percent duty cycle could be assured. Operation at a 33 percent duty cycle would yield an engine life of 18 hours with a molybdenum thrust chamber. If steady-state operation of the engine were desired, a reduction of specific impulse to 250 seconds at the 5-pound thrust level (or a mission specific impulse of approximately 235 seconds) would yield an engine with a life expectancy of 2 hours. Incorporating a reduction of specific impulse to 250 seconds in conjunction with a pulsing duty cycle of 50 percent would yield a tenfold increase in chamber life over a steady-state operation. Change in oxidizer-to-fuel ratio would not be recommended. Tantalum, which appears to have a higher temperature capability than molybdenum, must be investigated further through actual engine firings to verify its utility.

The thrust chamber wall temperatures were obtained in tests with the aluminide-coated 90 tantalum - 10 tungsten thrust chamber. During a 30-second run at the 5-pound thrust level the temperature in the highest temperature region reached 4260°F after 18 seconds of operation and decreased to 3780°F at the end of the 30-second run. The test results when compared to the temperatures obtained with the molybdenum disilicide coated molybdenum chambers apparently indicate that the aluminide coating has a lower emittance than coated molybdenum. This results in a much higher equilibrium wall temperature for an aluminide coated chamber than for a coated molybdenum thrust chamber. The test results to date indicate that an aluminide coated tantalum chamber cannot be considered for steady-state use in the Syncom program unless the outside surface emittance is increased to equal or exceed that for coated molybdenum.

Reliability. A reliability apportionment study of components, subsystem, and system has been completed and is being reviewed. A continuing review of Syncom polishing techniques on chambers is in process and a statistical analysis of the effect of variations of the polished surface on the chamber reliability is being initiated.

A 3-hour presentation on The Marquardt Corporation integrated Reliability and Quality Control was given at Hughes Aircraft Company.

In addition, a 3-day formal audit of TMC Reliability and Quality Control was completed.

Materials and Process. Design support included the following activities:

1) Consultation on nine drawing EOs.

2) Revision of Specification MMS-2204 (0. 5 titanium alloy) molybdenum bar to facilitate procurement and inspection.

3) Literature search on compatibility of candidate oxidizers.

4) Analysis of firing malfunction of altitude combustion chamber X18158-501, S/N 007. X-ray examination was made before and after coating and after firing. Metallurgical examination was made of cracks found after fire tests. Results indicate that the failure progressed inward from the outside diameter and may have resulted during test firing. Investigation is continuing.

5) Analysis of firing malfunction of combustion chamber S18158-501, S/N 006, was begun.

6) Submittal of materials and process input for Syncom II GSE proposal on system propellant decontamination.

7) The TMC IR&D sponsored altitude chamber of 90 tantalum - 10 tungsten was monitored during tin-aluminide coating at General Telephone and Electronics Research Laboratories at Bayside, Long Island. The outside diameter was spray coated in excess of 0. 0045 inch thickness. The inside diameter mix area was brush coated to 0. 0050-0. 0075 inch thickness. The throat coating thickness was 0. 002 inch. Chamber was fired on 8 June 1963. Evaluation is in progress.

Manufacturing support included monitoring the following activities:

1) Vendor disilicide coating process on first of two-cycle operation for S/N 008, 009, and 010.

2) Vendor brazing on three units of solenoid valve assembly P/N X-19176.

3) Vendor brazing on three units of injector head P/N X-19151.

4) Vendor heat treatment on three units of propellant tanks P/N X-19166, S/N 003, 004, and 005.

Quality control support included the following activities:

1) Coating thickness evaluation for chambers (X-19158-501), Z/N 008, 009, and 010. All specimens met drawing requirements of 0.0020 ±0.0003 inch.

2) Heat treatment evaluation for propellant tanks (X-19166) S/N 003, 004, and 005. All specimens met drawing requirements of minimum Rockwell C32.

3) Interpretation of X-ray films for weldments of propellant tanks (X-19166) S/N 003, 004, and 005. Films met requirements of specification MPS 1601 modified per drawing.

4) Monitoring hydrostatic pressure test of brazed injector head X-19151.

It was determined that the Nitrogen Division of Allied Chemical can supply N_2O_4 with a maximum water content of 0.01 percent at a premium price. This special grade of N_2O_4 compares with a 0.1 percent water limit for the standard specification material.

It is recommended that consideration be given to use of low-water-content N_2O_4 later in the program. While special procedures may be required when handling to assure that the water content remains low, potential system problems associated with corrosion would be minimized.

Manufacturing

Breadboard Assembly. The breadboard assembly is complete.

Engineering Model Structure Assembly. All components are completed and installed except for two motor assemblies and two X19236 variable restrictors. All components for the motors are complete except for flow checking the X19151 injector heads.

Test

Facilities and STE. All major facility and STE items are complete.

ATL Operations. Testing continued in the area. The use of manometer tube pulse flow meters proved successful in measurement of propellant flow during single-pulse engine operation.

Cell 9 Operations. A successful series of runs were made with the breadboard system in Cell 9. The breadboard model has been removed from the cell and necessary changes to accommodate the flight-type transducers used on the engineering model have been investigated. Facility revisions necessitated by these changes will be made during the engineering model setup period.

Product Support

All required materials and components for the fabrication and assembly of the engineering model, Phase I, propellant transfer equipment have been received. Fabrication of the GSE breadboard supports and manifolding is approximately 90 percent complete. Following fabrication, all equipment will be pressure checked, cleaned, and sealed until ready for use. A preliminary propellant transfer procedure has been drafted and is being reviewed by the Marquardt Safety Office.

Sun Sensor Analysis

A preliminary investigation of sun sensor assembly geometry, sensor output signal characteristics, and the design of the onboard electronics which utilize the sun sensor signals has shown the need for a more detailed description of sun sensor output signal characteristics than now exists. This detailed description will improve the accuracy of the on-board sunline angle measurements by allowing an optimum triggering level to be selected for the sun sensor circuits. It will also provide a more accurate estimation of the contribution to measurement errors made by mechanical misalignments, variations between individual sensors, and variations between sensor performance on earth and performance in space.

The shape of a typical sun sensor output pulse is shown in Figure 4-22. The offset angle, θ_T, at which the output pulse reaches some defined triggering level, V_T, is a function of the sun angle to the spin axis, sun intensity, and sensor construction tolerances. The exact relationship be between θ_T based on existing test data and assuming a triggering level of 0.100 volt is:

$$\theta_T = \frac{(0.75° \pm 25\%)}{\sin \phi_s} \qquad \text{for } \phi_s \text{ from 30 to 150 degrees}$$

A test program has been initiated on existing Syncom I spare and qualification units to aid in defining sensor performance. Tests in sunlight will be conducted to determine actual ψ and ψ_2 output pulse shapes for various values of ϕ_s (Figure 4-22). Tests on several single sensors will also be conducted to determine typical variations in pulse shapes between individual sensors. These tests should aid the preparation of a more accurate equation for θ_T as well as indicating the actual effects of this offset on the measurement of the angle, ϕ_s.

The existing specification on sun sensor performance has been expanded and is included in the Spacecraft Systems Design Section of this report under "Revisions and Changes to Subsystem Functional Performance Specification, dated April 1963".

TELEMETRY AND COMMAND

Telemetry and Command Antenna Design

A preliminary antenna configuration has been determined as a result of tests of the spacecraft mockup (Figure 4-23). It consists of 16 whip antennas mounted at equal angles about the periphery of the spacecraft at the rocket end. Alternate whips are radial and 30 degrees outward from the axial direction, the radial whips having a separate feed system from the other set (Figures 4-24 and 4-25).

Tests on a set of four whips driven 90 degrees apart showed approximately 10 db variation with spin at 136 mc as shown in Figure 4-26a. This variation was reduced to approximately 2 db with eight whips driven 45 degrees apart as shown in Figure 4-26b for the whips mounted at 30 degrees to the spin axis, and in Figure 4-26f for radial whips. The corresponding patterns at 148 mc are shown in Figures 4-26c and 4-26g. Patterns taken in the plane of the spin axis are shown in Figures 4-26d, e, h, and i. The coverage at the end of the satellite opposite the antenna end is satisfactory. The two sets of eight whips are mounted at different angles to reduce mutual coupling and give pattern diversity in the plane of the spin axis.

The spacecraft mockup is being prepared to test the complete 16-whip configuration. Tests to confirm the decoupling between the two sets of eight whips will be performed.

Telemetry Encoder

Design is progressing on approximately half of the telemetry encoder circuits.

Command Decoder

A detailed description of the proposed dual-mode command decoder follows, including a block diagram in Figure 4-27.

The preliminary design of the output amplifiers, the read tone interrupt, and the read tone present circuits is nearly complete. (The output amplifiers are necessary to implement interfaces between the command subsystem and other subsystems within the spacecraft.) The preliminary design of the reset and clock circuits and the layout of the diode matrix are well under way. A study of the filters used in Syncom I is being made to determine if they are suitable for use in Syncom II.

The proposed Syncom II command decoder is a three-tone, dual-mode system which, during operation, requires that only one tone be present at a time. The system requires only one input from the command receiver; it furnishes 128 individual outputs, each of which corresponds to a spacecraft command signal. A complete block diagram of the decoder is shown in Figure 4-27.

Under normal operating conditions the decoder responds in the primary mode to an input which is frequency-shift-keyed (FSK) modulated. The essence of this type of modulation is that the message is sent in with two tones, one to signify a logical one, the other a logical zero. The message is shifted into the command register with a bit sync clock signal which amplitude modulates both signal frequencies. Thus, while only one tone is present at any one time, the clock is always present.

The secondary mode of the decoder responds to a 100 percent pulse amplitude modulated input which corresponds to interrupting the message tone at regular intervals. This mode requires that only one tone channel be used to send the entire message. The command register counts the interrupts and so accumulates the desired message.

Decoder Operation

Prior to sending a message, both message tones must be stopped for a short period of time (<10 ms) during which timing circuitry is allowed to recover. It is assumed that one tone or the other has been on when a message is not being sent. This assumption may not necessarily hold, but it will if the command system is to be made secure from outside influences when it is not in use. If it is desired to operate in the primary mode (called shift mode), the zero tone is then turned on and left on until the message is to be sent.

Shifting to the ones tone will cause word sync to be detected, and the next seven bits will address enable power. Enable power corresponds to a latching switch which supplies power to the execute circuitry when enable is turned on. When it is off, the presence of an execute tone means nothing to the decoder. If the correct address is read after the seventh bit, enable power will come on. The eighth through fourteenth bits correspond to the command which will ultimately be executed, provided it is verified correctly by telemetry. After the fourteenth bit has been detected, the input to the command register is disabled and any further inputs to the decoder will be ignored by the command register. When verification is completed both message tones will be terminated and the execute tone sent, causing the command to be executed.

The operation of the secondary mode (count mode) is started in a manner similar to the primary mode. Both tones are interrupted for a short period of time, and then either tone is turned on and left on for some minimum length of time. The tone which has been present is then interrupted again, and, at this time, the decoder will shift into the secondary mode. After the tone has been absent for a period of time, either tone is sent with the required modulation. (The tone not being used will be off.) The command register will accumulate the enable address, and when the desired number of pulses has been sent the tone is interrupted again for a given length of time. During this time, the enable will read the contents of the command register, and if the address is correct, the enable will come on.

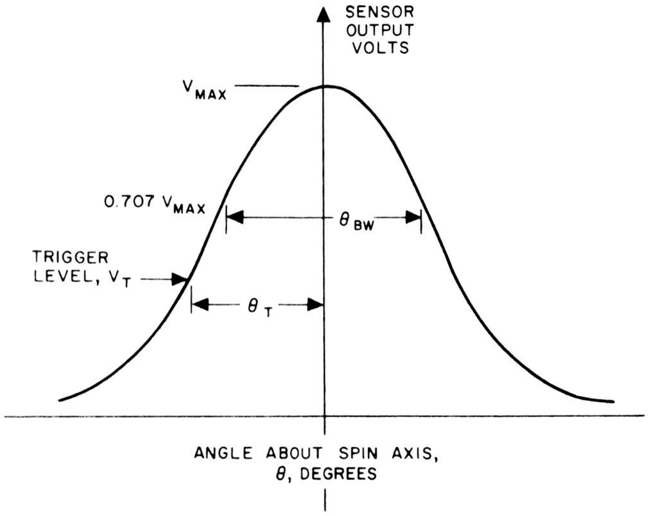


Figure 4-22. ψ and ψ_2 Sensor Output Pulse Shape

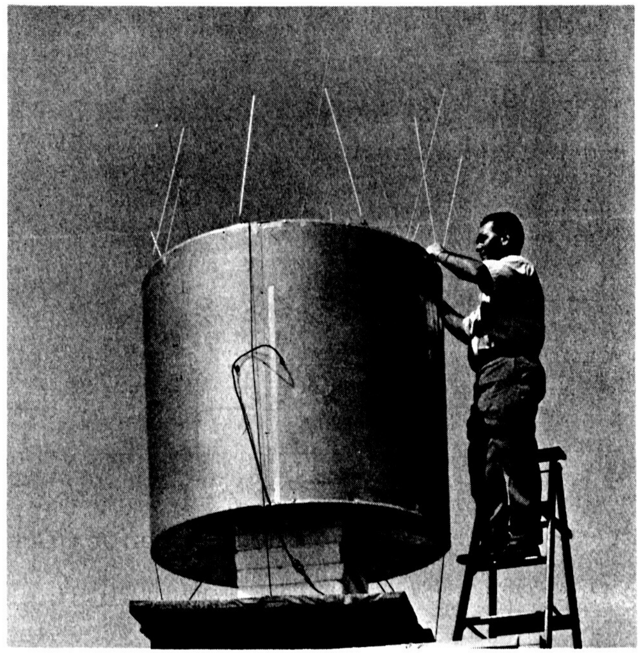


Figure 4-23. Spacecraft Mockup for Telemetry and Command Antenna Test

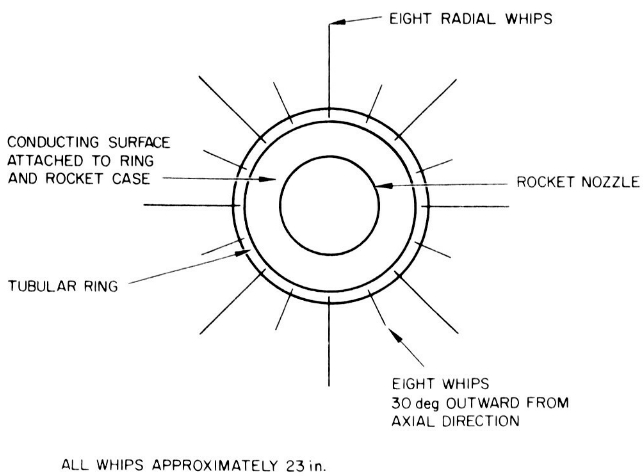


Figure 4-24. Nozzle End View Showing Whip Antenna Configuration

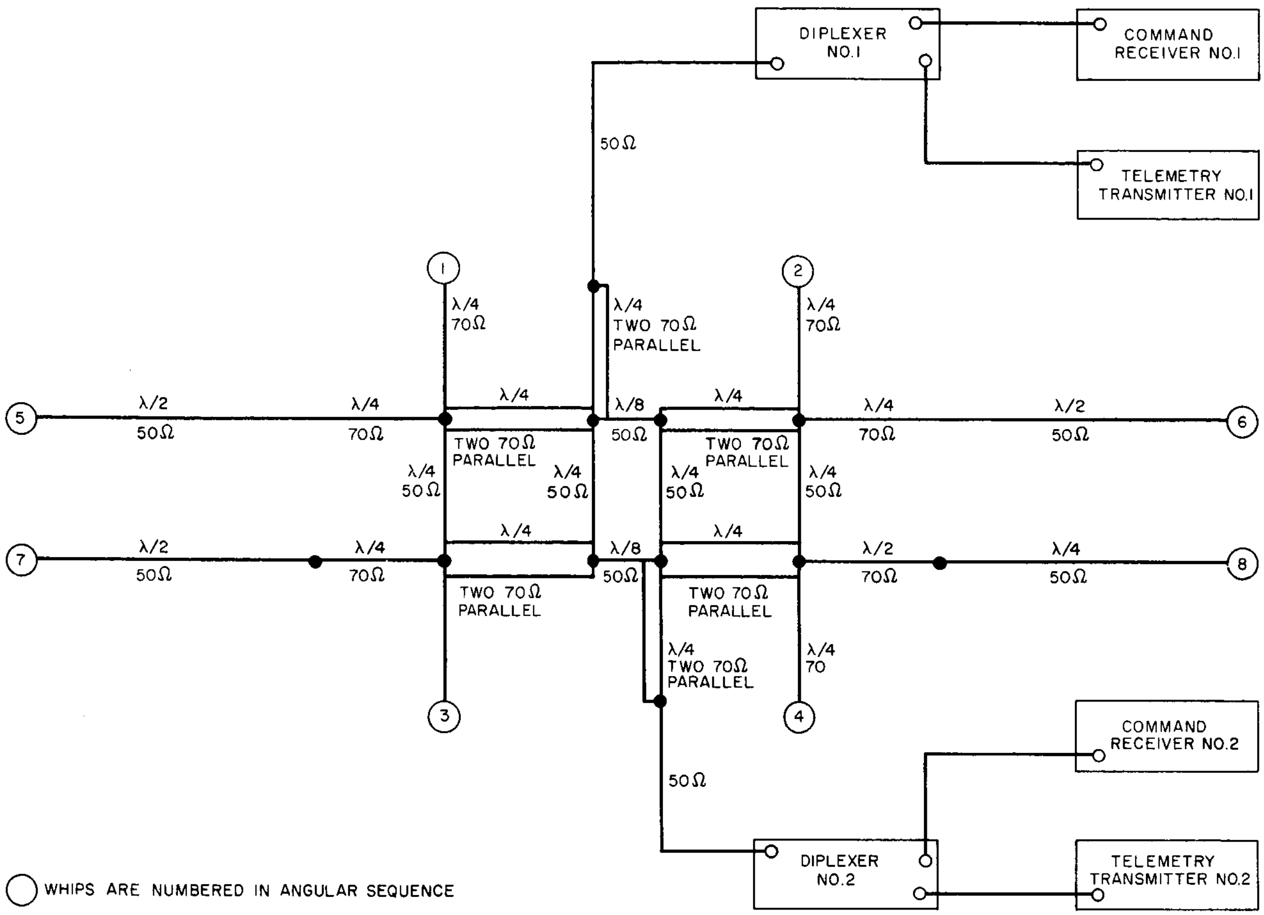
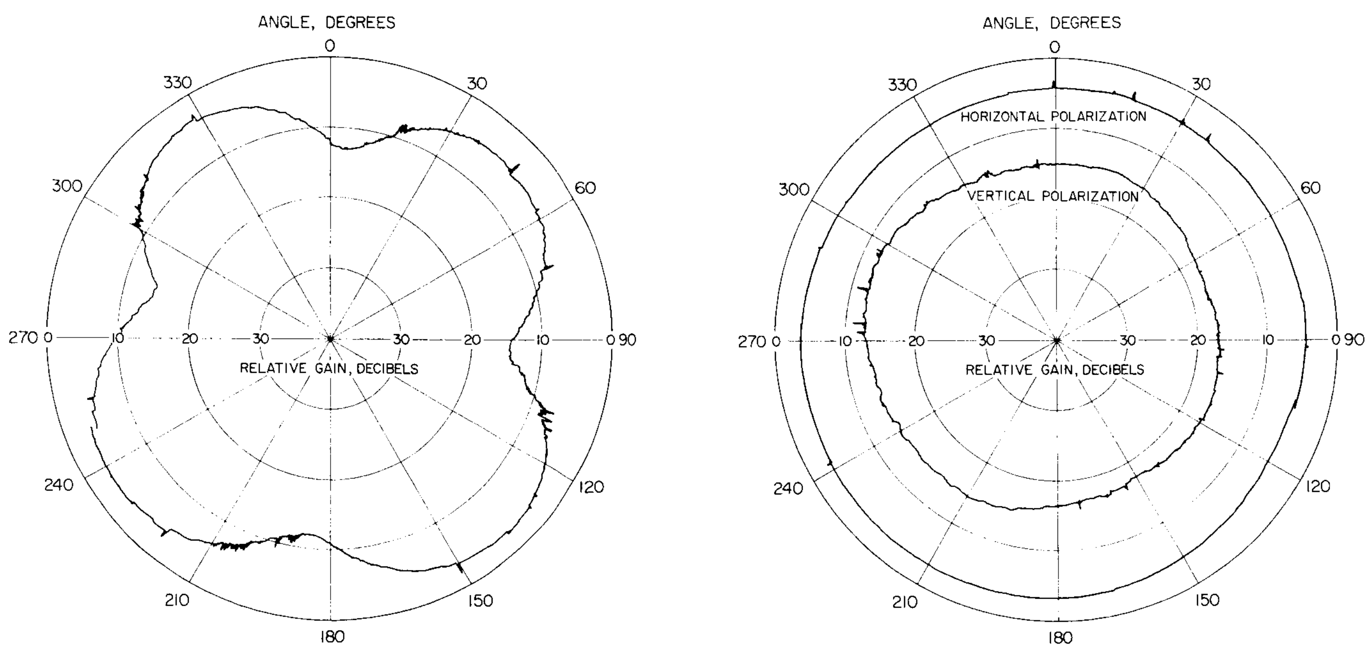


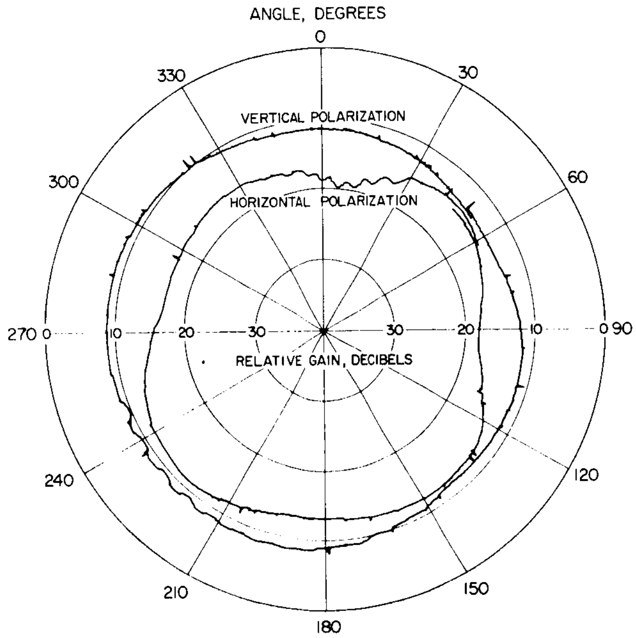
Figure 4-25. Feed System for Set of Eight Whips



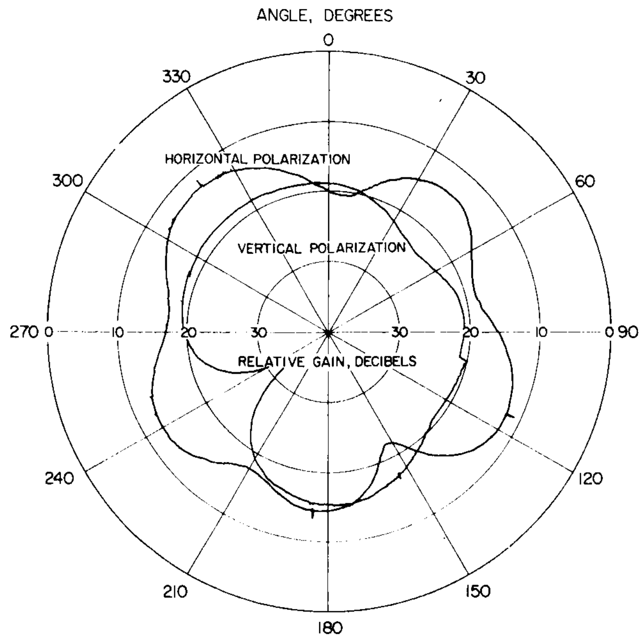
a) Four whips parallel to spin axis, 136 mc

b) Eight whips 30 degrees to spin axis, 136 mc

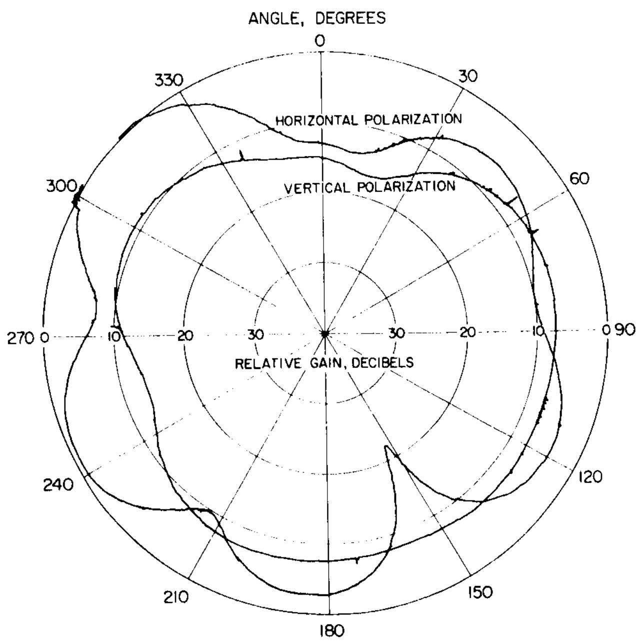
Figure 4-26. Telemetry and Command Antenna Pattern



c) Eight whips 30 degrees to spin axis, 148 mc

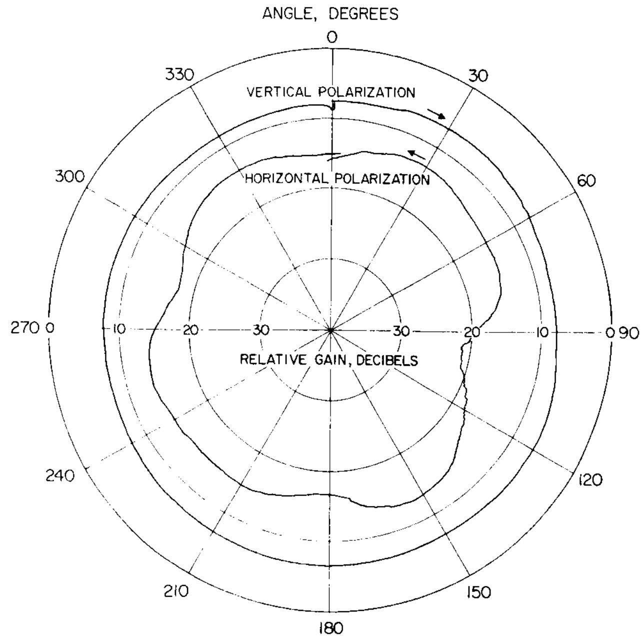


d) Eight whips 30 degrees to spin axis, 136 mc

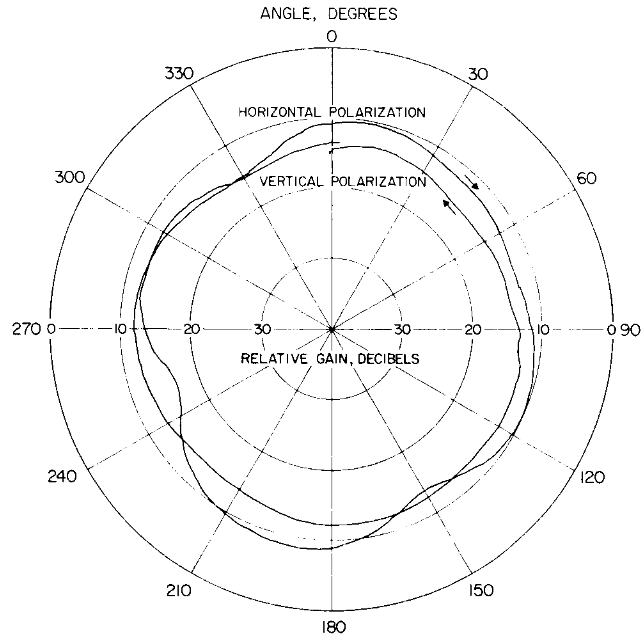


e) Eight whips 30 degrees to spin axis, 148 mc

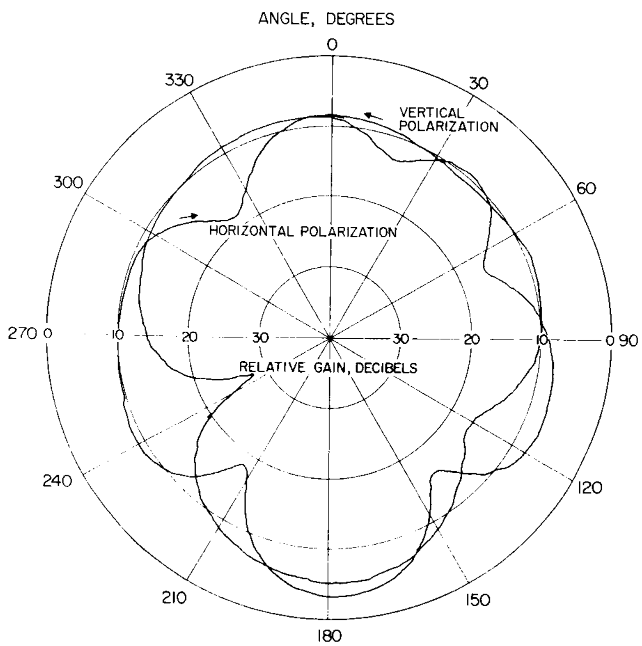
Figure 4-26 (continued). Telemetry and Command Antenna Pattern



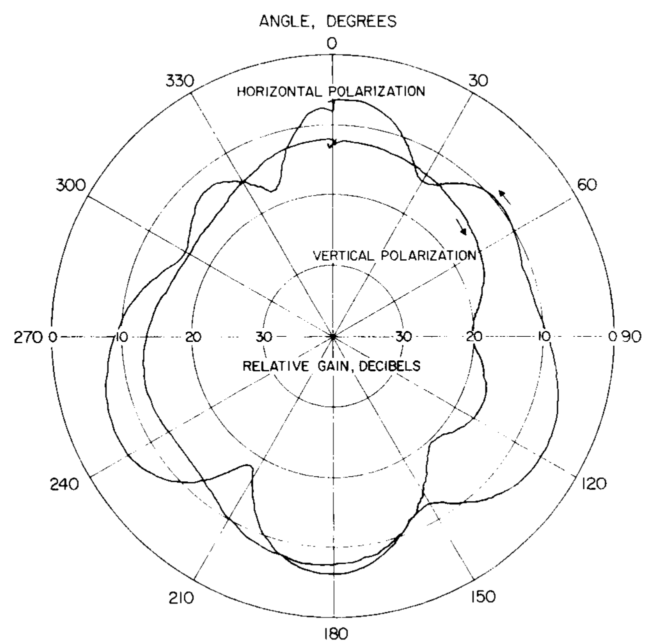
f) Eight whips 90 degrees to spin axis, 136 mc



g) Eight whips 90 degrees to spin axis, 148 mc (vertical)



h) Eight whips 90 degrees to spin axis, 136 mc



i) Eight whips 90 degrees to spin axis, 148 mc

Figure 4-26 (continued). Telemetry and Command Antenna Pattern

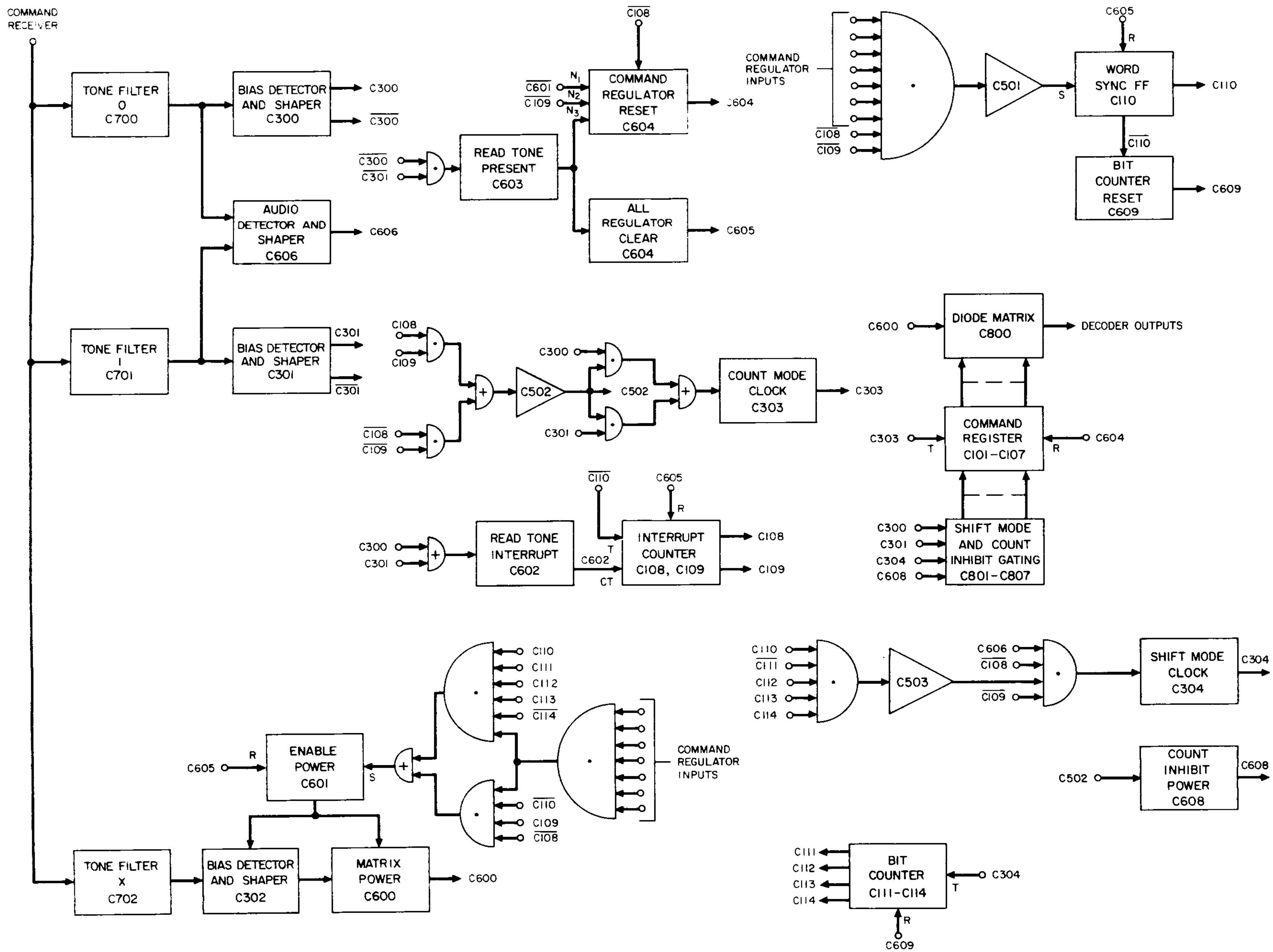


Figure 4-27. Command Decoder, Dual Mode

At the same time the command register will be reset to zero. The modulated input tone is then re-established and the command will be counted into the command register. After the desired command is accumulated the tone is turned off, and a short time later the input to the command register is disabled. Following verification by telemetry the execute tone is established, and the command is executed. Following execution of a command, the decoder can be completely cleared by applying a continuous uninterrupted message tone. This applies to either mode.

If, for either mode of operation, it is found that the contents of the command register does not correspond to the message sent from the ground, it is possible to clear the decoder completely. This is done by sending a continuous, uninterrupted message tone when in the count mode or by doing the same thing after interrupting both tones for a short period of time when in the shift mode.

Explanation of Functional Circuit Blocks. The nomenclature which is used below to identify the circuits described is defined as follows:

1) The C which precedes the number indicates that the circuit is part of the command decoder.

2) The first numeral identifies the type of circuit where:

1xx corresponds to a flip flop
3xx corresponds to an amplifier or shaper
5xx corresponds to an inverter
6xx corresponds to a circuit which is not a standard type
7xx corresponds to a filter
8xx corresponds to a special type of gating

3) The second and third numerals designate a particular circuit.

C700, C701, C702: The tone filters C700, C701, and C702 are narrow bandpass filters which will be designed to respond to three individual frequencies. C702 corresponds to the execute tone filter, C700 corresponds to what shall be called the zeros tone filter, and C701 corresponds to the ones tone filter. The input to the filters comes from the command receiver.

C300, C301, C302: The bias detectors and shapers C300, C301, and C302 have as their inputs the tone filters C700, C701, and C702 respectively. The bias detectors and shapers function as their name implies. They shape the outputs of the filters into waveforms which are suitable for digital use. They have a fixed bias level which, for a continuous sine wave amplitude modulated input, will put out a fixed unvarying voltage at some level greater than ground (+24 v). When a tone is completely absent the output shall be close to ground. When the input is 100 percent amplitude modulated with a pulsed type of waveform at a duty cycle which will be less than (preferred) or equal to 50 percent, the output will be pulsed also with an envelope which corresponds to the envelope of the input. (The latter case applies to the secondary mode.) The output of C302 will respond in a similar manner to C300 and C301 depending on the waveform of the execute.

C606: The audio detector and shaper, C606, has as inputs both C700 and $\overline{\text{C701}}$. The output is a square pulsed waveform regardless of the mode of operation. Waveform outputs of C300, C301, and C606 are shown in Figures 4-28 and 4-29 as a function of their inputs C700 and C701.

C603: The read tone present circuit, C603, has as its inputs $\overline{\text{C300}}$ and $\overline{\text{C301}}$. The output of C603 depends on the logical "AND" combination of the inputs. If both tone inputs to the decoder are absent long enough for the circuit to recover, the output will reach a logical "1" (high) state and stay there until a single tone (or the same logical function using both inputs, i.e., one always high while the other is low) has been present for a fixed length of time, and at this time the output will go to a logical "0" (low) state with a fast fall time.

C604, C605: The reset circuits C604 and C605, called respectively the "command register reset" and "all register clear" circuits, are triggered by negative voltage transitions at the input. The output is a short pulse which goes slightly negative during the pulse interval and resets the desired individual flip-flops or registers of flipflops. C604 is triggered by C603 (as is C605) and also by negative transitions of C108 (when C109 is low) and enable power, C601, which are discussed below.

C602: The read tone interrupt circuit, C602, is controlled by the logical "OR" combination of C300 and C301. When either tone has been present for some period of time such that C602 has time to recover to the logical "1" state, then the absence of both tones for a reasonably long period of time (after the recovery period) will cause C602 to fall to logical "0" with a fast fall time.

C108, C109: The interrupt counter, C108 and C109, consists of flip-flops. C108 is a clocked pedestal gated flip-flop and C109 is a standard pedestal gated flip-flop the input of which is the flip-flop output of C108. The "T" input to C108 is controlled by $\overline{\text{C110}}$ which is discussed below. The clock for C108 is derived from C602. As the name implies, the interrupt counter counts the negative transitions of C602, which correspond to interrupts in the input tone provided $\overline{\text{C110}}$ is in the logical "1" state. The interrupt counter is reset to zero by C605 which is controlled by C603 (discussed above).

C110: The word sync flip-flop, C110, is used to detect word sync when no interrupts have occurred prior to detection. When C108 and C109 are low and the proper configuration comes up in the command register the inverter C501 will set C110 in the logical "1" state indicating that word sync has been detected. C110 is reset to zero by C605.

C609: The bit counter reset, C609, resets the bit counter with a short duration pulse which goes to a negative voltage during the pulse interval. The pulse is initiated by a negative transition of C110.

C111, C112, C113, C114: The bit counter, C111 to C114, is made up of four pedestal gated flip-flops. It is used to count the bits which are being shifted into the command register after word sync has been detected. (Since the bit counter is reset to zero by $\overline{\text{C110}}$ it is ready to count from zero after word sync has been detected.)

C303, C304: The count mode clock, C303, and the shift mode clock, C304, are simple "AND - OR" amplifiers which are controlled by input gating. When the decoder is in the primary mode, C303 is in the low state, and C304 provides the shift clock. When the decoder is in the secondary mode, C304 is in the low state, and C303 provides the count clock. A more detailed discussion of the control gating is given below in the detailed mode description.

C601: The enable power circuit, C601, is a latching switch which provides power to bias detector and shaper C302, and the matrix power (C600) circuits. The enable power circuit is turned on by a correct address present in the command register at the proper time, and it is reset by C605 (discussed above).

C101, C107: The command register, C101 - C107, consists of seven flip-flops which can, through the use of digital switches and special gating, be used to accumulate a desired message by counting or shifting in the input.

C608: The count inhibit power circuit, C608, is used to provide power to gating which prevents incongruities from occurring in the command register, when operating in the shift mode, due to the presence of the count mode circuitry.

C800: The diode matrix, C800, is an arrangement of diodes which provides all possible combinations (128) of the binary outputs of the command register. The power to the gating resistors of the matrix is controlled by C600 designated the matrix power circuit.

The operation of C302 was discussed above. The output is high when an execute tone is present provided that the enable has been turned on.

C600: The matrix power circuit, C600, works in conjunction with C302 in that it provides power to the diode matrix when an execute tone is present provided that enable power is on. When an execute tone is present all of the gates in the matrix will have power applied to them and the required command signal will be distributed throughout the spacecraft.

Detailed Decoder Operation. The explanation of decoder operation can more easily be accomplished by reference to the timing waveforms in Figures 4-28 and 4-29, which apply to the shift and count mode respectively.

It is assumed, regardless of the mode of operation to be chosen, that initially one message tone (as opposed to the execute tone) will be on, although this may not necessarily be the case.

Primary (Shift) Mode Operation. Output waveforms discussed below are referenced to Figure 4-28. Prior to sending a message to the spacecraft, the tone which has been on must be interrupted for several milliseconds. Following this period of time the zeros tone is turned on and the output of C700 will appear as shown between t_1 and t_2. A minimum of 15 bit sync cycles must elapse before the word sync is sent. The total interrupt of the message tone allows C603 to recover, and between t_1 and t_2 the fall in the output of C603 resets all of the decoder registers to zero with the exception of the bit counter C111 to C114. Operation in this mode, after t_2, never allows C603 to recover again, since henceforth a tone will always be present at one input or the other.

At the time, t_2, the states of the mode control circuits in the decoder will be as follows. The logical configuration of the interrupt counter, C108 and C109 will cause C502 to be low. As a result, C608 will be providing power to the count inhibit circuitry, and C303 (count mode clock) will be disabled. The shift mode clock, C304, will be operating, and zeros will be shifted into the command register until word sync is detected.

Between t_2 and t_3 several ones are sent, and when the right number is detected C501 will go low (at t_3), setting the word sync flip-flop, C110, high. When C110 goes high, the bit counter will be reset to zero by C609 and the input to the interrupt counter will be disabled. After t_3 the bit counter will count the number of clock pulses put out by the shift mode clock, C304.

Starting at t_3 the command register begins to accumulate the enable address, and at t_4, seven bits after the detection of word sync, provided the address is correct, enable power will be turned on by the bit counter in combination with the contents of the command register.

At t_4 the command register begins to accumulate the command and as soon as the seven command bits have been accumulated, the shift clock, C304, will be turned off and its output will stay at ground. This occurs at t_5, i.e., fourteen bit times after the detection of word sync, and the shift clock is disabled.

As soon as verification (via the telemetry link) is complete, the message tone, if it is on, is turned off and the execute tone is turned on. C302 will then turn on the matrix power circuitry and the command will be executed within the spacecraft (provided that enable power is on).

After the execution technique is completed, either tone can be turned on and the command register will be cleared so that undesired executions cannot occur accidentally. Also, the decoder will be secure against any possible outside influences.

Secondary (Count) Mode Operation. Output waveforms discussed below are referenced to Figure 4-29. As in primary mode operation, the continuous tone which has been present for some time is interrupted and C603 is allowed to recover. At time, t_1, either tone can be turned on, and sometime later all of the registers in the decoder again, with the exception of the bit counter, will be reset to zero. The interrupt time is of the same order of magnitude as that for the primary mode. At time, t_2, the input tone is interrupted again, and a short time later (at t_3) C602 goes low setting C108 high.

When C108 goes high, the input to the word sync flip-flops, C110, is disabled and C110 will remain in the reset condition. Also, when C108 goes high, the output of C502 will go high disabling the count inhibit power circuitry and enabling the input to the count mode clock, C303. The input to the shift mode clock will be disabled at the same time, and its output will stay at ground. The command register will also be reset to zero by C604.

At t_3 the decoder is ready to count short interrupts in the input tone. When the tone is reinstated it will be pulse amplitude modulated and the command register will count the number of pulses until time, t_4, when the enable address has presumably been counted into the command register. At this time the input is interrupted again for a reasonably long period of time, and C602 will go low causing C108 to go low and C109 to go high. This will occur at t_5, and at this time the combination of C109, C108, and the command register output configuration will cause enable power to come on provided the correct address is present in the command register. When the enable power comes on, the command register will be again reset to zero by C604.

After the command register has been reset at t_5 the pulse amplitude modulated input tone is resumed and the command is counted into the command register. When the right number of counts has been accumulated the tone is again interrupted and a short time later C602 will make another negative transition causing C108 to go high and so disabling the count mode clock and causing it to remain near ground. Following verification the command is executed as in the primary mode.

ELECTRICAL POWER

Simplified Power System Design

During the May NASA Design Review, a possible simplification of the electrical power system, particularly in battery charging circuitry, was discussed.

The system is schematically portrayed by Figure 4-30. This system involves the isolation of an area of each of the 16 spacecraft solar panels to charge the batteries directly. No battery charging regulator for current

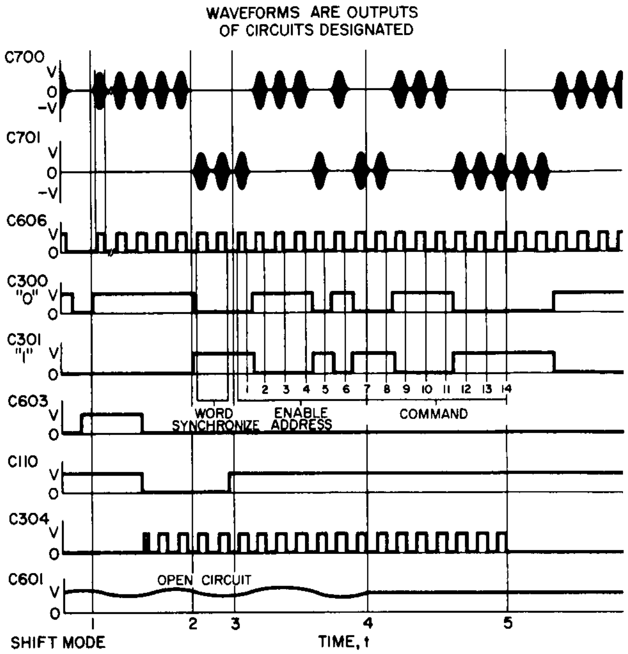


Figure 4-28. Command Decoder, Shift Mode

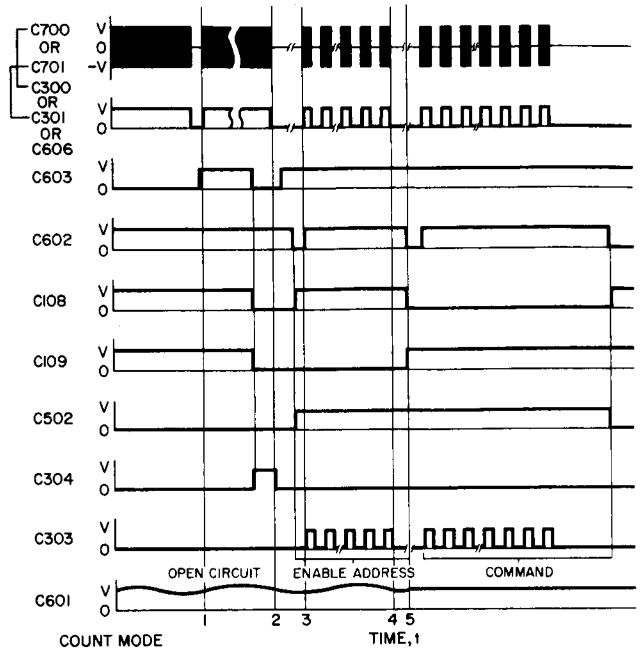


Figure 4-29. Command Decoder, Count Mode

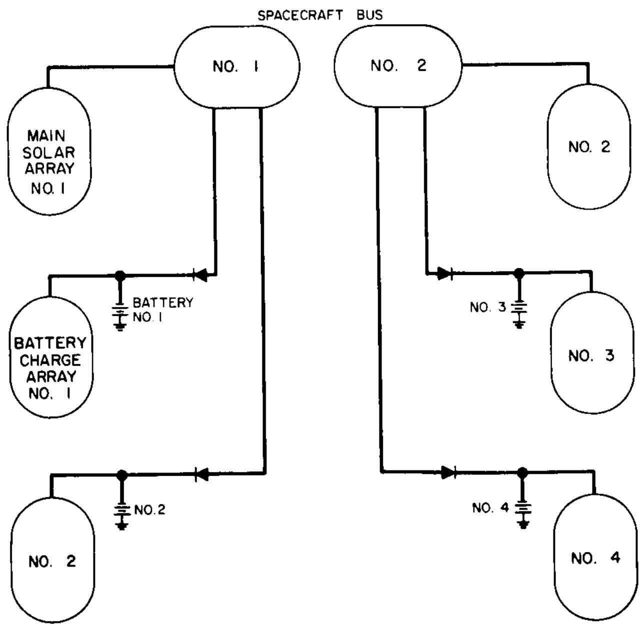


Figure 4-30. Simplified Schematic of Power System

limiting in flight would be required since a solar array in itself is a fairly precise current limiter. The battery charging portion of the array would be one of the three parallel string groups of each panel (eight groups per panel). Alternate panel groups would be connected to each battery. A diode (similar to Syncom I) would be used as a simple logic device to connect the battery to the bus to provide power as required. Any excess current from the battery charging portion of the array not needed for battery charging would be delivered to the bus via the diode. The primary items where the end result is lowering of the solar array voltage are as follows:

1) Random solar cell failures

2) Solar flare activity

3) Van Allen radiation

4) Solar cell temperature rise

The power loss of the Advanced Syncom as a result of the above considerations is summarized in Table 4-17.

Table 4-17

PERCENT OF POWER LOSS IN SOLAR ARRAY VOLTAGE

Time in Orbit Years	Temperature Increase	Random Failure Rate	Solar Flares		Van Allen Radiation		Total Radiation	
			6 mil Covers	30 mil Covers	6 mil Covers	30 mil Covers	6 mil Covers	30 mil Covers
1	0.5%/°C	5.94	19	9	30	7.5	34	10.5
2	0.5%/°C	8.1	24	12	36	10	36	13.5
3	0.5%/°C	11.2	27	14	40	11.5	40	16.0
4	0.5%/°C	13.5	29	16	43	13	43	17.5
5	0.5%/°C	15.5	30	18	45	14	45	18.0

Table 4-17 is derived from the latest information available on both the Syncom orbit as well as solar cell degradation as a function of both electron and proton irradiation. It is obvious from the above that the use of 6 mil cover glasses should not be considered further. The above degradations are plotted in Figures 4-31a thru 4-31d.

Simulated Syncom II Battery Cycling Sequence

One 22-cell nickelcadmium battery (Syncom I) composed of Sonotone 1/2 C cells has been placed on a 24 hour test cycle. The charge-discharge values used (Figure 4-32) are ratios of the Syncom I and II battery capacities and the proposed Syncom II loads. The loads of Syncom II are assumed to be approximately 1 ampere per battery during eclipse operation. The ambient temperatures are as shown in Figure 4-33. The minimum battery temperature is about 15 degrees above the chamber ambient due to thermal lag and cell internal losses under discharge. The battery ampere-hour efficiency has been assumed to be 45 percent. The test data available to date is summarized on Figure 4-32.

In an effort to more closely simulate the Advanced Syncom conditions, 22 ADVENT cells (12 ampere-hour) are being placed on the 24 hour test cycle to more closely approximate the actual vehicle system.

Fabrication Status of Battery Strings

The revised battery specifications, which are included in this section, have been released for initiation of cell procurement. It is contemplated that cells will be available for test during the latter part of August.

170262 Solar Cell Procurement Specification

1.0 Scope

1.1 This specification covers the requirements for the design and construction of a photovoltaic solar cell to be used on the Synchronous Communications Satellite (Syncom) II solar panel assembly.

1.2 Design Objectives. The solar cell shall be designed to meet all electrical, optical, mechanical, and environmental requirements as specified herein. Test programs shall be successfully completed demonstrating the ability of the solar cell to meet all performance requirements as required by this specification. The solar cell shall be designed for optimum operation in accordance with the following relative priority list:

1) Reliability
2) Air mass zero sunlight conversion efficiency
3) Spectral characteristics
4) Thermal characteristics
5) Weight

1.3 Conflicting Requirements. Conflicting requirements arising between this specification or of any specification or drawing listed herein shall be referred in writing to Hughes Aircraft Company for interpretation and clarification.

1.3.1 Requests for Deviation. Requests for deviation from this specification, applicable drawings, specifications, publications, materials, and processes specified herein, shall be considered design changes or design deviations and shall not be allowed except by written authorization from Hughes.

1.4 Materials, Parts and Processes. When a material, part, or process is not specified herein, the seller's selection shall assure the highest uniform quality and condition of the product, suitable for the intended use, and such selection shall be submitted for the review and concurrence of Hughes.

2.0 Applicable Documents

2.1 The following documents of the date and/or revision shown are a part of this specification except as noted in subsequent paragraphs:

Military Specifications

MIL-STD-105	Sampling Procedures and Tables for Inspection by Attributes

Hughes Aircraft Company Specifications

225001	Quality Assurance, 17 May 1961

NASA Specifications

63-106	Specification for Determining Relative 1 Mev Electron Radiation Damage Resistance for Silicon Solar Cells, 31 October 1962

3.0 Requirements

3.1 Design Description. The solar cell shall be three ohm cm^{-1} silicon "N" on "P" junction type, 1 by 2 centimeter size, the coverglass applied. The individual solar cells shall be covered with a coverglass capable of meeting the requirements specified herein. The solar cells will be capable of being bonded to a honeycomb substrata, and will be capable of being electrically interconnected in series-parallel groups to assure maximum reliability of operation.

3.1.1 Configuration: The dimensions and overall configuration of the solar cell shall be specified in the seller's drawing and shall be submitted for Hughes approval.

3.1.2 Cell Defects: The maximum chip allowed shall be 0.010 inch deep by 0.150 inch long and the maximum corner crack shall be 0.045 inch on the hypotenuse.

3.1.2.1 Cell Covers: Each solar cell shall be covered with a 0.030 inch thick Corning Number 7940 Silica Quartz coated with an anti-reflection coating on the top surface and an ultraviolet reflecting coating on the under surface. Cell to cover overlap and exposed active area shall not be greater than 0.005 inch. Density shall be 2.2 grams cm^{-3}.

3.1.2.2 Cell Cover and Adhesive Defects: Cracks, scratches or discoloration will not be allowed. Chips will not extend more than 0.010 inch from an edge. There shall be no evidence of delamination, discoloration or bubbles in the adhesive. Covers shall not have bubbles detectable by the unaided eye.

3.1.2.3 Solar Cell Absorptance and Emittance: Average emittance of the solar cell top surface with the coverglass applied shall not be less than 0.83 from 25°C to 125°C. Average absorptance to solar radiation in the wavelength region 0.2 to 2.5 microns shall not exceed 0.82.

3.1.2.4 Spectral Transmittance: The spectral transmittance of the cell cover shall meet the following requirements:

Wavelength, Microns	Transmittance, Percent
0.400 ± 0.015	50
From 0.300 to 0.370	Less than 1 average
From 0.500 to 1.000	92 minimum

3.1.3 Negative Contact: The exposed negative (top) contact of the cell shall have a clean, uniform, and complete line of solder along the 2 cm dimension. The width of this solder line shall not be less than 0.032 inches.

3.1.4 Positive Contact: The positive contact of the cell shall be flat within 0.005 inch and free of all contaminating material. The surface shall be optimized so as to provide maximum adhesive bonding strength.

3.1.5 Contact Coverage: Solder coverage of "N" and "P" contacts shall be 90 percent minimum.

3.1.6 Weight: Total assembled solar cell weight including coverglass shall not exceed 0.577 grams average per lot.

3.2 Power Output. The power output of each solar cell with coverglass applied under air mass zero spectral conditions and solar radiation intensity of 140 mw/cm^2 shall meet the following requirements under test conditions:

Temperature	25°C (±2°)
Voltage	0.45 volts (±0.010v)
Power	22.7 milliwatts (minimum)

The specified voltage shall be used in measuring the power output of the solar cells. The electrical performance of the solar cell shall be measured with an illumination source as specified in Paragraph 3.2.1.

3.2.1 Illumination Source: The source of radiation used to illuminate the cell for purposes of confirming cell power Paragraph 3.2 shall be sunlight at the earth's surface at Table Mountain, California, or at other Hughes-approved test sites with the following minimum sunlight conditions:

1) 100 mw/cm^2 illumination intensity

2) Five miles clear visibility

3) Minimum sky radiation. This shall be determined by the ratio of solar cell short circuit current under the conditions of uncollimated and collimated sunlight. The ratio shall be as follows:

$$\frac{I_{sc} \text{ (uncollimated sunlight)}}{I_{sc} \text{ (collimated sunlight)}} \leq 1.08$$

A collimating tube equipped with baffles shall be used and the tube shall have a minimum length to diameter ratio of 10.

The power output under the test conditions of Paragraph 3.2 and at 100 mw/cm^2 intensity shall be multiplied by the factor 1.21 to obtain the air mass zero power output. The test data obtained for each cell subjected to test shall be submitted to Hughes concurrent with delivery of each lot.

3.2.2 Temperature Variations: The seller shall furnish the voltage-current characteristics curves of the solar cell for 0°, 75° and 125°C. The tests shall be run with a constant illumination source as specified in Paragraph 3.2.1 or equivalent.

3.2.3 Illumination Intensity Variations: The seller shall furnish the voltage-current characteristic curves of the solar cell at intensities of 100, 115, 130 and 150 milliwatts per square centimeter.

3.3 Storage. The solar cell as specified in Paragraph 3.1 with coverglass installed, shall be capable of meeting the requirements specified below:

1) The solar cell shall be capable of meeting all performance requirements after storage at a relative humidity of 50 percent maximum and at a temperature of 21 ± 15°C for a period of 24 months.

2) The solar cell shall be capable of meeting all performance requirements after storage at a relative humidity of 95 percent maximum and at a temperature of 24 ± 20°C for a period of one month.

3.4 Radiation Damage Resistance. The seller shall provide evidence of compliance with the NASA-GSFC Radiation Damage Procurement Specification, Specification No. 63-106, dated 31 October 1962.

3.5 Environmental Performance. The solar cell shall meet all performance requirements of this specification after having been subject to the environmental conditions specified in Paragraph 4.0 of this specification.

3.6 Interchangeability. Solar cells bearing the same part number shall be physically and functionally interchangeable without selection or fit. The Hughes part number for this solar cell shall be 170263.

4.0 Tests

4.1 General

4.1.1 Test Apparatus: All meters, scales, thermometers, and similar measuring test equipment used in conducting tests specified herein shall be accurate within 1 percent of the full-scale value. Full-scale deflections of meters should not be more than twice the maximum value of the quantity being measured. All test apparatus shall be calibrated at suitable intervals and records of such calibration shall be available for inspection by Hughes. Hughes may examine the seller's test equipment and determine that the seller has available and utilizes correctly gauging, measuring, and test equipment of the required accuracy and precision, and that the instruments are of the proper type and range to make measurements of the required accuracy. The calibration of gauges, standards, and instruments will be checked in a mutually agreed upon primary standards laboratory if disputes concerning performance occur.

4.1.2 Test Records: Records shall be kept of all tests and of applicable manufacturing data, and these records shall be made available for inspection by Hughes. Prior to and following each test of Paragraph 4.6, a thorough visual examination of the test solar cell shall be conducted. All physical markings, defects, and other visual characteristics shall be noted and recorded as a portion of the test records.

4.1.3 Test Conditions: Unless otherwise specified herein, all tests shall be performed at the following nominal ambient conditions:

1)	Temperature	25°C
2)	Barometric pressure	29.92 inches of mercury
3)	Relative humidity	not greater than 50%

4.2 Classification of Tests. Tests shall be classified as follows:

1) Acceptance Tests
2) Type Approval Tests

4.3 Sampling Procedure. The sampling procedure for acceptance tests of Paragraph 4.5 shall meet the requirements of Military Specification MIL-STD-105 for an AQL of 2.5% defective, excluding the Electrical Performance Tests of Paragraph 4.5.2.

4.4 Test Location. Unless otherwise specified in the contract, type approval and acceptance tests shall be performed by the seller at the seller's plant. If the use of outside test facilities are required, the use of these facilities shall be subject to approval by Hughes. Hughes shall have the right to witness, inspect, and review all type approval and acceptance tests.

4.5 Acceptance Tests. Samples of all lots of solar cells submitted for delivery shall be subjected to the acceptance tests listed below. A lot shall consist of 25,000 solar cells manufactured under essentially the same conditions and submitted for acceptance at substantially the same time. The sampling plan shall comply with Paragraph 4.3.

4.5.1 Examination of Product: The solar cell shall be inspected to determine compliance with respect to materials, workmanship, dimensions, and weight as specified in Paragraphs 3.1.1, 3.1.2, 3.1.3, 3.1.4, 3.1.5, and 3.1.6.

4.5.2 Electrical Performance: The power output of the solar cell shall be determined at a temperature of 25° ± 2°C. To comply with the requirements of Paragraph 3.2, 200 solar cells out of each 25,000 solar cell lot will be selected in a random manner and their electrical performance determined in sunlight as specified in Paragraph 3.2.1. The data obtained from this sunlight measurement will be employed to calibrate a laboratory light source and thereby establish acceptance criteria for the solar cells at the seller's and buyer's facility. In addition to seller's acceptance tests, electrical performance tests of delivered solar cells will be conducted by Hughes. Any cell determined to be defective during Hughes inspection shall be cause for rejection of the entire lot. The light source used by the seller for the above testing shall have the approval of Hughes.

4.6 Type Approval Tests. Type approval tests shall be conducted in the manner described below and prior to final contract award. A sample of 100 solar cells with coverglass shall be selected at random from a production lot. When one or more test samples fails to meet the requirements of this specification, the extent and cause of failure shall be determined and corrective action initiated. After corrective action has been taken, type approval and acceptance tests shall be repeated as mutually agreed between Hughes and the seller upon review of the failure analysis. All cells subjected to type approval tests shall not be used for flight hardware. The solar cells shall be subjected to type approval tests in the order listed below.

4.6.1 Acceptance Tests: All solar cells shall be tested in accordance with and meet the requirements of Paragraph 4.5.

4.6.2 Electrical Performance Test: The power output of the solar cells shall be measured in accordance with Paragraph 3.2. Temperature of the solar cells shall be continuously monitored. The solar cells shall meet the requirements of Paragraph 3.2.

4.6.3 Storage Temperature and Humidity: The test specimens shall be placed in a sealed test chamber and the temperature and humidity raised during a 2-hour period to 52°C and 95 percent relative humidity, respectively. At the end of a 6-hour soak period, the heat source for the chamber will be turned off. During the following 16-hour period, the temperature shall drop at a uniform rate to 37°C or less. Three such 24-hour cycles shall be performed consecutively. At the end of this period, electrical performance tests in accordance with Paragraph 4.6.2 shall be conducted.

4.6.4 Temperature Cycling: The solar cells shall be subjected to five temperature cycles at a minimum thermal rate of 30°C per minute, between the extremes of 110° ± 2°C and -196° ± 2°C. The solar cells shall remain at the extremes for a minimum of 1 hour. Electrical performance tests in accordance with Paragraph 4.6.2 shall then be conducted.

4.6.5 High Temperature--Vacuum: The solar cells shall be placed in a test chamber and the chamber reduced in pressure to a vacuum of at least 10^{-5} Torr. The temperature shall be raised to 110° ± 2°C. The solar cells shall remain in the chamber for a period of 168 hours. At the end of this period, the solar cells shall be allowed to return to room ambient temperature and the electrical performance tests in accordance with Paragraph 4.6.2 shall be conducted.

4.6.6 Ultraviolet Radiation Tests: Fifteen of the 100 type approval solar cells shall be subjected to high intensity ultraviolet radiation from a Model No. 700-J Ultra-Violet Lamp Unit manufactured by Shannon Luminous Materials Company, Hollywood, California, or the equivalent. If the Shannon Lamp Unit is employed, no more than eight cells at a time shall be irradiated. The cells shall be positioned normal to the irradiation with the active cell areas facing the illuminating source. The cells shall be positioned about the centerline of the lamp unit at a distance of approximately 3 1/2 inches from the open end of the lamp housing. Forced air cooling shall be employed to maintain the cells at a temperature in the range 40° to 50°C. Duration of the test shall be 20 hours. Upon completion, the cells shall be tested for electrical performance in accordance with Paragraph 4.6.2.

4.6.7 Paragraph 3.1.2.2 shall apply after each test in Paragraphs 4.6.3, 4.6.4, 4.6.5 and 4.6.6.

4.7 Radiation Damage. In order to comply with Paragraph 3.4, the seller shall conduct the radiation damage tests in accordance with NASA-GSFC, Specification No. 63-106 or the seller shall provide sufficient evidence these tests have previously been completed satisfactorily. This test will not be considered a part of the type approval program but must also be completed prior to final contract award.

4.8 Retest. Any changes made in manufacturing techniques, processes, materials, quality control levels, manufacturing sites or type of manufacturing equipment shall be cause for complete retest per Paragraph 4.6 at no cost to Hughes.

5.0 Preparation for Delivery

5.1 Shipping Container. The seller shall provide containers of the size required for the delivered lots with a desiccant capable of assuring container ambient relative humidities of no greater than 50 percent in compliance with the requirements of Paragraph 3.3.1.(1). Desiccant may be replaced periodically if necessary. An indicator of desiccant water absorption should be provided.

5.2 Identification. Each solar cell shipping box shall be legibly identified by the following:

1) HAC part number
2) Seller's part number
3) Month and year of manufacture
4) Lot number
5) Solar cell serial number (1 through 25,000 for each lot)

6.0 Quality Assurance Provisions

6.1 General. The materials, processes, and assembly covered by this specification shall be subject to extensive inspection and testing by both the seller and Hughes.

6.2 Inspection

6.2.1 Seller Inspection: The seller shall establish a quality control system in accordance with or exceeding the requirements of Hughes Specification 225001, Quality Assurance Specification. Product quality assurance shall be provided by the seller by a series of in-process inspections commencing with receipt of raw materials and parts and continuing through the finished product. The selected inspection points shall have the approval of Hughes. A record shall be maintained of all inspections and be subject to review by Hughes.

6.2.2 Hughes Source Inspection: The Hughes Aircraft Company shall at its option provide inspection to adequately monitor the seller's quality control effort including in-process inspection and in-process tests. The completed hardware may be source inspected by Hughes to assure that the product conforms to all the requirements specified on the applicable drawings and specifications and may include witnessing of acceptance tests.

6.2.3 Rejected Assemblies - Rejected assemblies shall not be resubmitted for approval without furnishing full particulars concerning the rejection, the measures taken to overcome the defects, and the prevention of their future occurrence. Each rejected assembly shall be identified by a serialized rejection tag. This rejection tag shall not be removed until rework requirements have been complied with, and then the tag shall be removed only by, or in the presence of, an authorized representative of Hughes.

X30630-001 Battery Cell Procurement Specification

1.0 Scope

1.1 This specification covers a hermetically sealed nickel-cadmium battery cell to be used in the assembly of batteries for space applications.

3.0 Requirements

3.1 Design Description. The cell shall be hermetically sealed nickel-cadmium type suitable for space application as specified herein.

3.1.1 Weight: The weight shall not exceed 0.65 pounds.

3.1.2 Terminals: All electrode terminals shall be insulated from the case and contain provisions for solder-type connections of the lead wires.

3.1.3 Container: The cell container shall be capable of maintaining its original dimensions for the life of the battery under the storage and operating conditions specified herein.

3.1.4 Corrosion Resistance: All external surfaces of the cell shall show no evidence of corrosion when exposed to the environmental conditions specified herein.

3.1.5 Leakage: The cell shall show no signs of electrolyte leakage when subjected to the storage and operating conditions specified herein. The cell shall show no signs of leakage when tested in accordance with Paragraph 4.3.2.

3.1.6 Interchangeability: All cells having the same part number shall be functionally and dimensionally interchangeable.

3.1.7 Cell Marking: The following information shall be marked by stamping, etching or other suitable methods which will insure permanent legibility:

1) Hughes Aircraft Company part number

2) Serial number

3) Manufacturer's name, trademark, or code symbol

4) Terminal identification

3.2 Performance Requirements

3.2.1 Capacity: The cell discharge capacity at 75°F shall be a minimum of 6.0 ampere-hours when discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The voltage for 3.5 hours of the discharge period shall be 1.16 volts minimum.

The charge schedule to determine the ampere-hours discharge capacity and voltage requirements of this paragraph and paragraphs 3.2.2 through 3.2.5 shall be a constant current charge at 0.5 amperes for 16.0 hours followed by an open-circuit period of 1.0 hour. The charge shall be from a point of previous discharge to 1.0 volts at 1.2 amperes.

3.2.2 Capacity at Low Temperature: With the cell case temperature maintained at 30°F during charging and discharging, the discharge capacity shall not be less than 4.8 ampere-hours when charged and discharged for the periods and rates specified in Paragraph 3.2.1.

3.2.3 Capacity at High Temperature: With the cell case temperature maintained at 120°F during charging and discharging, the discharge capacity shall not be less than 4.8 ampere-hours when charged and discharged for the periods and rates specified in Paragraph 3.2.1.

3.2.4 Capacity at High Rate Discharge: The cell discharge capacity at 75°F shall be a minimum of 4.8 ampere-hours when discharged at a constant current of 6.0 amperes to an end voltage of 1.0 volts.

The charge schedule used to meet the requirements of this paragraph shall be the same as the charge schedule specified in Paragraph 3.2.1.

3.2.5 High Current Discharge: The cell terminal voltage during discharge of a fully-charged cell at a load of 12.0 amperes shall be 1.0 volts minimum for a period of 10 seconds.

3.2.6 Overcharging Rate: With the cell case temperature maintained at 75°F, the cell shall be capable of withstanding a continuous overcharging current 0.5 amperes for a period of 30 days.

3.2.7 Maximum Charge Voltage: The maximum on-charge cell voltage shall not exceed 1.48 volts when meeting the requirements of Paragraph 3.2.6.

3.2.8 Charge Retention: The cell discharge capacity in ampere-hours, shall not be less than 80 percent of its initial capacity when discharged 30 days after being fully charged.

The cell shall meet the provisions of this paragraph when charged and discharged at the rates and periods specified in Paragraph 3.2.1. During the 30 days stand time, the cell case temperature will be maintained at 75°F.

3.2.9 Charge Retention at Minimum Charge: The cell open-circuit voltage shall be 1.16 volts minimum after 24 hours stand time when tested in accordance with Paragraph 4.3.4.

3.2.10 Charging at Minimum Rate: When charged at a constant current of 0.060 amperes for a period of 180 hours at 75°F, the charge discharge capacity shall not be less than 5.4 ampere-hours. The capacity shall be measured by discharging at a constant current of 1.2 amperes to an end voltage of 1.0 volts. Prior to discharge, the cell shall stand on open-circuit for a period of one hour.

3.2.11 Capacity After Cycling: The cell discharge capacity shall be a minimum of 4.8 ampere-hours after 500 charge-discharge cycles (at 75°F) of one hour discharge at 1.2 amperes and 7.0 hours charge at 0.3 amperes. In addition, the discharge voltage shall be a minimum of 1.20 volts during the 500 discharge cycles.

3.3 Environmental Requirements

3.3.1 Storage: Each cell shall be capable of meeting all the requirements of this specification after storage for two years at any temperature between 20°F and 130°F.

3.3.2 Vacuum: Each cell shall be capable of meeting all the requirements of this specification in a vacuum environment of at least 10^{-10} mm Hg.

3.3.3 Humidity: Each cell shall be capable of meeting all the requirements of this specification after being subjected to a test chamber temperature of 130°F and a relative humidity of 95 percent for 8 hours.

3.3.4 Thermal Shock: Each cell shall be capable of meeting all the requirements of this specification after being subjected to a test chamber temperature of -20°F for at least 6 hours immediately followed by exposure to a test chamber temperature of 150°F for at least 6 additional hours.

3.3.5 Shock: Each cell shall be capable of meeting all the requirements of this specification after being subjected to two 60-g terminal peak sawtooth shock pulses of 15 millisecond duration each in each direction along the three principal cell axes. During each shock, the cell shall be capable of being charged and discharged in accordance with any of the requirements of Paragraph 3.2 with no significant voltage change due to the shock.

3.3.6 Acceleration: Each cell shall be capable of meeting all the requirements of this specification after being subjected to steady accelerations of 30 g for 60 seconds duration in each direction along the three, principal cell axes. While being accelerated, the cell shall be capable of being charged and discharged in accordance with any of the requirements of Paragraph 3.2 with no significant change in voltage due to the acceleration.

3.3.7 Spin: Each cell shall be capable of meeting all the requirements of this specification while being spun continuously in any attitude at 140 rpm from a 26-inch radius.

3.3.8 Vibration: Each cell shall be capable of meeting all the requirements of this specification after being subjected to the vibration environment listed below along the three principal cell axes. During vibration, the cell shall be capable of being charged and discharged in accordance with any of the requirements of Paragraph 3.2 with no significant change in voltage due to the vibration.

1) Sinusoidal Excitation

Frequency, cps	Log Sweep Rate, Octaves/ Minute	Duration, Minutes	Level
5 to 15	2.0	4.3	0.25 in. double amplitude
15 to 250	2.0	4.3	3.0 g (0 to peak)
250 to 400	2.0	4.3	5.0 g (0 to peak)
400 to 2000	2.0	4.3	7.5 g (0 to peak)
50 to 80	1.0	4.3	7.5 g (0 to peak)

2) Random Excitation

Frequency, cps	Duration, Minutes	Level
20 to 80	6.0	0.04 g^2/cps
80 to 1280	6.0	Increasing from 0.04 g^2/cps at 1.22 db/octave
1280 to 2000	6.0	0.07 g^2/cps

4.0 Tests

4.1 General

4.1.1 Test Apparatus: All meters, scales, thermometers, and similar measuring test equipment used in conducting tests specified herein shall be accurate within 1.0 percent of the full scale value. Full scale deflections of meters should not be more than twice the maximum value of the quantity being measured. Periods of discharge and charge shall be timed with a device accurate within 0.2 percent. All test apparatus shall be calibrated at suitable intervals against standards traceable to the National Bureau of Standards. Records of such calibration shall be available for inspection.

4.1.2 Records: Records shall be kept and be made available for inspection of the tests and of applicable manufacturing data (e.g., serial numbers of batteries manufactured from each lot of raw or processed material).

4.1.3 Test Conditions: Unless otherwise stated, laboratory ambient conditions of tests shall be:

1)	Temperature	70 ± 10°F
2)	Barometric pressure	30 ± 2 inches of Mercury
3)	Relative humidity	less than 90 percent

4.1.4 Tolerances: Unless specifically stated in the test procedures, the following test tolerances are allowable:

1)	Ambient temperature	± 5°F
2)	Relative humidity	± 5 percent
3)	Vibration level	± 10 percent
4)	Pressure	± 5 percent
5)	Frequency	± 2 percent
6)	Shock	± 10 percent
7)	Acceleration	± 10 percent

4.1.5 Rejections and Retest: When one or more cells from a lot fails to meet the requirements of this specification in a manner indicative of a systematic design deficiency, acceptance of all items in the lot will be withheld until the extent and cause of the failure is determined and corrective action initiated. A lot shall consist of cells manufactured essentially under the same conditions, from the same materials stock, and at the same time. After corrective action has been taken, acceptance and qualification tests shall be repeated as mutually agreed between Hughes and the cell manufacturer upon review of the failure analysis. Cells, which have been rejected, may be reworked or replaced to correct any defects and re-submitted for acceptance. Before re-submitting the cells for test, full particulars concerning the rejection and corrective action taken shall be furnished to Hughes. If investigation of a test failure indicates that defects may exist in cells already accepted, these cells shall be retested and reworked or replaced as required to comply with this specification. Cells which fail to meet specific selection or acceptance test requirements shall be rejected on an individual cell basis.

4.1.6 Additional Tests: Additional tests shall be conducted by Hughes as deemed necessary to verify that the cell can meet the requirements of this specification. These tests shall not impose more stringent requirements than those specified in this for rejection in accordance with Paragraph 4.1.5.

4.2 Classification of Tests. Tests shall be classified as follows:

1) Acceptance Tests.

2) Qualification Tests.

4.3 Acceptance Tests. All cells submitted for delivery shall be subjected to the following tests. These tests shall be conducted at laboratory ambient conditions. Upon completion of each test, specimens and test data shall be examined to determine compliance with this specification.

4.3.1 Examination of Product: Each cell shall be inspected to determine compliance with respect to material, workmanship, dimensions, weight, and product marking.

4.3.2 Leakage Test: A leakage test shall be conducted on each cell by one of the two methods described below:

1) Helium Leak Tests (for cells containing Helium gas). The cell shall be placed in a vacuum chamber and the pressure reduced to at least 1 x 10^{-5} mm Hg and maintained for at least 5 minutes. The helium leakage rate from the cell shall be measured with a Consolidated Electrodynamic Corporation, Model 24-120 leak detector or equivalent. The leakage rate shall not exceed 1 cubic centimeter of helium per month.

2) Electrolyte Indicator Leak Test. In lieu of the helium leak detection method, an electrolyte indicator test may be used. The indicator shall be a one percent solution of phenolphthalein in alcohol or P-H indicator paper. Any change in color of the indicator shall be evidence of electrolyte leakage.

4.3.3 Capacity Discharge Test: Each cell shall be charged per Paragraph 3.2.1, allowed to stand on open circuit for 1 hour, and then discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volts. This test shall be repeated following the test of Paragraph 4.3.5 for a second charge-discharge cycle. Each cell shall meet the discharge capacity and voltage requirements of Paragraph 3.2.1 during each cycle.

4.3.4 Charge Retention, Minimum Charge: Following the discharge of Paragraph 4.3.3, the cell shall be short circuited for 12 hours minimum. The short circuit shall be removed and the cell charged at a constant current of 0.5 amperes for 10 minutes. The cell shall then be placed on open circuit for 24 hours during which time the open circuit voltage of the cell shall be 1.16 volts minimum (see Paragraph 3.2.9).

4.3.5 Overcharge: Following the test of Paragraph 4.3.4, the cell shall be charged at 0.5 amperes for a period of 96 hours. The cell on-charge voltage shall not exceed 1.48 volts. Following this test on the cell shall again be subjected to the leakage test and meet the requirements of Paragraph 4.3.2.

4.4 Qualification Tests. Battery cells submitted for qualification tests shall be typical of production line batteries of the final design for flight usage. A minimum of 20 cells shall be subjected to the tests of Paragraph 4.4.1 through 4.4.13 and a minimum of 20 cells for the test of Paragraph 4.4.1 through 4.4.5 and 4.4.14. The tests on each cell shall be conducted in the order listed below. All cells subjected to qualification testing shall meet all requirements herein.

4.4.1 Acceptance Tests: All cells submitted for qualification tests shall be tested in accordance with and meet the requirements of Paragraph 4.3.

4.4.2 Thermal Shock Test: The cells, after being fully charged, shall be placed in a vacuum chamber maintained at 1×10^{-5} mm Hg. The cells shall then be subjected to a test chamber temperature of -20°F for a period of 6 hours followed immediately by exposure to a test chamber temperature of + 150°F for an additional 6 hours. Open circuit voltage of each cell shall be 1.25 volts minimum following this test (see Paragraph 3.3.4).

4.4.3 Vibration Test: The cells after being fully charged shall be mounted rigidly to a test fixture and subjected to the vibration environment in each of the three orthogonal axes as shown below. During each mode of vibration the cells discharged at 1.2 amperes. Cell voltages shall not change by more than 0.01 volts due to the vibrations (see Paragraph 3.3.8).

1) Sinusoidal Excitation

Frequency, cps	Log Sweep Rate, Octaves/Minutes	Duration, Minutes	Level
5 to 15	2.0	4.35	0.25 in. double amplitude
15 to 250	2.0	4.35	3.0 g (0 to peak)
250 to 400	2.0	4.35	5.0 g (0 to peak)
400 to 2000	2.0	4.35	7.5 g (0 to peak)
50 to 80	1.0	4.35	7.5 g (0 to peak)

2) Random Excitation

Frequency, cps	Duration Minutes	Level
20 to 80	6.0	0.04 g^2/cps
80 to 1280	6.0	Increasing from 0.04 g^2/ cps at 1.22 db per octave
1280 to 2000	6.0	0.77 g^2/cps

4.4.4 Shock Test: The cells after being fully charged shall be mounted rigidly in test fixture and subjected to two 60-g terminal peak saw-tooth shock pulses of 15 milliseconds duration in each direction along the three principal cell axes. During the shock, the cells shall be subjected to a 1.2 ampere discharge and show no change in voltage greater than 0.01 volts due to the shock (see Paragraph 3.3.5).

4.4.5 Acceleration Test: The cells after being fully charged shall be subjected to steady accelerations of 30 g for 60 seconds duration in each direction along the three principal cell axes. During the accelerations, the cells shall be subjected to a 1.2 ampere discharge and show no change in voltage greater than 0.01 volts due to the acceleration (see Paragraph 3.3.6).

4.4.6 Spin Test: Following the test of Paragraph 4.4.5, each cell shall be discharged to 1.0 volts. The cells shall then be spun in a test fixture at 208 rpm from a radius of 26 ± 1 inch. The cells shall be oriented so that the longitudinal axis of the cell is along the radius vector of rotation and the cell terminals face the center of rotation. While spinning the cells shall be charged at 0.5 amperes for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. Each cell shall meet the discharge capacity and voltage requirements of Paragraph 3.2.1 (see Paragraph 3.3.7).

4.4.7 Low Temperature Capacity Test: Following the test of 4.4.6, the cells shall be placed in a temperature chamber and maintained at 30°F throughout this test. The cells shall be charged at 0.5 amperes for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall not be less than 4.8 ampere-hours for each cell (see Paragraph 3.2.2).

4.4.8 High Temperature Capacity Test: Following the test of Paragraph 4.4.7, the cells shall be placed in a temperature chamber and maintained at 120°F throughout this test. The cells shall be charged at 0.5 amperes for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall not be less than 4.8 ampere-hours for each cell (see Paragraph 3.2.3).

4.4.9 Capacity at High Rate Discharge: Following the test of Paragraph 4.4.8, the cells shall be placed in a laboratory temperature environment of 75°F. The cells shall be charged at 0.5 amperes for 16.0 hours, placed on open circuit for 1.0 hour and then discharged at a constant current of 6.0 amperes to an end voltage of 1.0 volts. The discharge capacity shall not be less than 4.8 ampere-hours for each cell (see Paragraph 3.2.4).

4.4.10 Charging at Minimum Rate Test: Following the test of Paragraph 4.4.9, the cells shall be charged at a constant current of 0.060 amperes for a period of 200 hours, placed on open circuit for 1 hour and then discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall not be less than 5.4 ampere-hours (see Paragraph 3.2.10).

4.4.11 High Current Discharge Capability Test: Following the test of Paragraph 4.4.10, the cells shall be fully charged, placed on open circuit for 1 hour, then discharged at a rate of 12.0 amperes for a period of 10 seconds. The voltage for the 10-second discharge period shall be 1.0 volts minimum (see Paragraph 3.2.5).

4.4.12 Overcharge Test: Following the test of Paragraph 4.4.11, the cells shall be continuously overcharged at a constant current of 0.5 amperes for a period of 30 days. During this period the cells shall be in a 75°F temperature environment. The cell on-charge voltage shall not exceed 1.48 volts during the 30-day period (see Paragraph 3.2.6).

4.4.13 Charge Retention Test: Following the test of Paragraph 4.4.12, the basic capacity of the cells shall be redetermined in accordance with the test procedure of Paragraph 4.3.3. The cells shall then be charged to full capacity and placed on open circuit for a period of 30 days. At the end of 30 days, the discharge capacity shall not be less than 80 percent of the initial discharge capacity measured just prior to the 30-day period (see Paragraph 3.2.8).

4.4.14 Cycle Test: After the cells have completed the tests of Paragraphs 4.4.1 through 4.4.5, they shall be subjected to the following cycle testing:

Cycles	Charge Current	Charge Time	Discharge Current	Discharge Time	Ambient Temperature
1 - 500	0.3 amps	7.0	1.2 amps	1.0 hour	75°F

At the end of each 100 cycles, the basic ampere-hour capacity to 1.0 volts shall be determined in accordance with the test procedure of Paragraph 4.3.3. At the end of 500 cycles, the discharge capacity shall not be less than 4.8 ampere-hours. In addition, the end-of-discharge voltage during the 500 cycles shall not be less than 1.2 volts (see Paragraph 3.2.11).

X30630-002 Battery Cell Procurement Specification

1.0 Scope

1.1 This specification covers a hermetically sealed nickel-cadmium battery cell which contains a sensory oxygen electrode to indicate when the cell reaches a fully-charged state. This cell will be used in series with cells described in Hughes Procurement Specification No. X30630-001 in the assembly of batteries for space vehicle usage.

2.0 Applicable Documents

Hughes Procurement Specification No. X30630-001, Sealed Nickel-Cadmium Battery Cell, 6.0 Amp.-Hr.

3.0 Requirements

In addition to the requirements of this specification, the cells shall meet all the requirements of HAC Specification X30630-001 without utilizing the oxygen electrode.

3.1 Performance Requirements of the Sensory Electrode

3.1.1 Electrical Output: With the cell being charged at any rate between 0.060 amperes and 0.5 amperes and a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the indication that the cell has reached a fully-charged state shall be an increase of potential difference between the oxygen electrode and negative electrode to 0.9 volt minimum. During charging, when the cell is in a state less than fully charged, the potential difference between the oxygen electrode and the negative electrode shall be less than 0.3 volts.

3.1.2 Minimum Cell Performance at Maximum Full Charge Indication: The requirements of Hughes Aircraft Company Specification X30630-001, Paragraphs 3.2.1, 3.2.2, 3.2.3, and 3.2.10 shall also be met by charging the cell at the prescribed rate for each test until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volt. At this point the charge shall be terminated and the cell tested in accordance with the remaining provisions of the paragraph. This requirement shall be met with a maximum external resistance of 1.5 ohms between the oxygen electrode terminal and the negative electrode terminal.

3.1.3 Minimum Cell Performance at an Intermediate Charge Indication: The cell shall meet the ampere-hour requirements of Hughes Aircraft Company Specification X30630-001, Paragraphs 3.2.1, 3.2.2, 3.2.3, and 3.2.10 diminished by 10 percent, by charging the cell at the prescribed rate for each test until the potential difference between the oxygen electrode and the negative electrode reaches 0.3 volt. At this point the charge shall be terminated and the cell tested in accordance with the remaining provisions of the paragraph. This requirement shall be met with a minimum external resistance of 25 ohms between the oxygen electrode terminal and the negative electrode terminal.

3.1.4 Charge Indication During Discharge: During discharge from a fully-charged state, the 0.9 volt indication between the oxygen electrode and the negative electrode shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged. This requirement shall be met with a minimum external resistance of 25 ohms between the oxygen electrode terminal and the negative electrode terminal. This requirement shall be met coincidentally with ampere-hour discharge requirements of Hughes Aircraft Company Specification X30630-001, Paragraphs 3.2.1, 3.2.2, 3.2.3, and 3.2.10.

4.0 Tests

In addition to the tests of this specification, the cells shall be subjected to and meet the test requirements of Sections 4.1, 4.2, 4.3, and 4.4 of Hughes Aircraft Company Procurement Specification X30630-001 as specified herein.

4.1 Acceptance Tests. In addition to the Acceptance Tests of Section 4.3 of Hughes Procurement Specification X30630-001, the following tests shall be performed on all cells submitted for delivery.

4.1.1 Cell Capacity Test at Full Charge Indication (75°F): With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 amperes constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volts. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volts. This test shall be repeated for a second charge-discharge cycle. Each cell shall demonstrate a discharge capacity for each cycle of 6.0 ampere-hours. In addition, the voltage for 3.5 hours of the discharge period shall be 1.16 volts minimum. This test shall be performed in a 75°F environment.

4.2 Qualification Tests. Four cells typical of production line batteries of the final design for flight use shall be subjected to all the qualification tests of Paragraph 4.4 of Hughes Procurement Specification X30630-001, except for Paragraph 4.4.14. In addition, the same cells shall be subjected to the qualification tests of Paragraph 4.2.1 through 4.2.8 of this specification. Four additional cells shall be subjected to the qualification tests of Paragraphs 4.4.1 through 4.4.6 of Hughes Procurement Specification X30630-001. In addition, the same cells shall be subjected to the qualification test of Paragraph 4.2.9 of this specification.

4.2.1 Cell Capacity Test at Full Charge Indication (30°F): This test shall be performed in a test chamber maintained at 30°F. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 amperes constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volts. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for 1 hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volts. The discharge capacity shall be 4.8 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.2 Cell Capacity Test at Full Charge Indication (120°F): This test shall be performed in a test chamber maintained at 120°F. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 amperes constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volts. At this point the charge shall be terminated and the cell allowed to stand on open circuit for 1 hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volts. The discharge capacity shall be 4.8 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.3 Cell Capacity Test at Full Charge Indication at Minimum Charge Rate: This test shall be performed at 75°F ambient temperature. With a 1.5 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.060 amperes constant current until the potential difference between the oxygen electrode and the negative electrode of the battery reaches 0.9 volts. At this point the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall be 5.4 ampere-hours minimum. In addition, the 0.9 volt indication shall decrease to 0.3 volt before 15 percent of the ampere-hour capacity is discharged.

4.2.4 Cell Capacity Test at Intermediate Charge Indication (75°F): With a 25-ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 amperes constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.3 volt. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for 1 hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volt. Each cell shall demonstrate a discharge capacity for each cycle of 5.4 ampere-hours minimum. This test shall be performed in a 75°F environment.

4.2.5 Cell Capacity Test at Intermediate Charge Indication (30°F): This test shall be performed in a test chamber maintained at 30° F. With a

25 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 amperes constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.3 volts. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for 1 hour. The cell shall then be discharged at constant current of 1.20 amperes to an end voltage of 1.0 volts. The discharge capacity shall be 4.3 ampere-hours minimum.

4.2.6 Cell Capacity Test at Intermediate Charge Indication (120°F): This test shall be performed in a test chamber maintained at 120°F. With a 25 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 amperes constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.3 volts. At this point, the charge shall be terminated and the cell allowed to stand on open circuit for 1 hour. The cell shall then be discharged at a constant current of 1.20 amperes to an end voltage of 1.0 volts. The discharge capacity shall be 4.3 ampere-hours minimum.

4.2.7 Cell Capacity Test at Intermediate Charge Indication and Minimum Charge Rate: This test shall be performed at 75°F ambient temperature. With a 25 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.60 amperes constant current until the potential difference between the oxygen electrode and the negative electrode of the battery reaches 0.3 volts. At this point the charge shall be terminated and the cell allowed to stand on open circuit for one hour. The cell shall then be discharged at a constant current of 1.2 amperes to an end voltage of 1.0 volts. The discharge capacity shall be 4.9 ampere-hours minimum.

4.2.8 Charge Indication During Discharge (Maximum Impedance): With a 25 ohm external resistance between the oxygen electrode and the negative electrode of the battery, the cell shall be charged at 0.5 amperes constant current until the potential difference between the oxygen electrode and the negative electrode reaches 0.9 volts. At this point the charge shall be terminated and the cell shall be discharged immediately at 1.20 amperes to an end voltage of 1.0 volts. During discharge the 0.9 volt indication between the oxygen electrode and the negative electrode shall decrease to below 0.3 volt before 15 percent of the ampere-hour capacity of the cell is discharged.

4.2.9 Cycle Test: After the cells have completed the tests of Paragraph 4.4.1 through 4.4.6 of the Hughes Procurement Specification X30630-001, they shall be subjected to the following cycle testing:

Cycles	Charge Current	Charge Time	Discharge Current	Discharge Time	Ambient Temperature
1 to 500	0.3 amps	to 0.9V indication on sensory electrode	1.2 amps	1.0 hour	75°F

At the end of each 100 cycles the ampere-hour capacity shall be determined in accordance with the test procedure of Paragraph 4.1.1. At the end of 500 cycles, the discharge capacity shall not be less than 4.8 ampere-hours. In addition, the end-of-discharge voltage during the 500 cycles shall not be less than 1.2 volts.

Solar Array Reliability Assessment

The system reliability studies have included a detailed formulation of the expected solar array degradation due to random failures occurring during the orbital lifetime of the spacecraft. A technique for determining the distribution of failures in a multi-redundant solar cell configuration is described and applied to the Syncom II solar array.

In general, a solar array may be described as consisting of solar cell strings arranged in series-parallel groups to provide the required current output and voltage level for the spacecraft equipments. In a multi-redundant solar array the strings are interconnected to form alternate series-parallel current paths such that each individual cell failure would not cause the loss of the output of a complete string. Thus, the multi-redundant configuration may be defined as consisting of a number of solar cell groups each with m rows of cells and n cells per row with each cell interconnected in a series-parallel-grid arrangement. Individual cells are assumed to exhibit the exponential failure distribution with failures independently and randomly distributed about the solar array.

The approach to the analysis will be to determine the expected failure distribution occurring in an individual cell group and to apply the result to the array.

The probability of survival (reliability) of an individual cell is defined as

$$P_{cell} = e^{-\lambda_c t}, \tag{4-1}$$

where λ_c is the cell failure rate, and t is the orbital mission time.

The probability of failure of an individual cell (q_{cell}) is one minus the probability of survival, or

$$q_{cell} = 1 - p_{cell} = 1 - e^{-\lambda_c t} . \tag{4-2}$$

The probability of failure of r cells within a row of the group may be expressed as the binomial probability distribution:

$$P(r) = \binom{n}{r} q^r p^{n-r} . \tag{4-3}$$

The probability of s rows with exactly r failures is the binomial probability distribution

$$P(s) = \binom{m}{s} P(r)^s Q(r)^{m-s} , \tag{4-4}$$

which may be approximated by the poisson probability distribution

$$P(s) \approx \frac{\left(mP(r)\right)^s e^{-mP(r)}}{s!} , \tag{4-5}$$

when $mP(r)$ is large relative to $P(r)$ and m is large relative to $mP(r)$. Note: $Q(r) = 1-P(r)$.

The probability that at least one cell row will have exactly r failures is one minus the probability of having no cell row with r failures:

$$P(\text{at least one cell row with exactly r failures}) = 1 - P(s = o)$$
$$= 1 - Q(r)^m \cong 1 - e^{-mP(r)} \tag{4-6}$$

This becomes the desired expression for the probability that a multi-redundant series-parallel cell group will have at least one cell row with exactly r failures.

Multiplying this expression by N for the number of multi-redundant cell groups which comprise the solar array and varying r yields the expected number of cell groups, E(g), having at least one cell row with exactly r failures for any orbital time, t, and cell failure rate, λ_c.

$$E(g) = N\left[1 - Q(r)^m\right] \approx N\left[1 - e^{-mP(r)}\right] \tag{4-7}$$

The technique described may be applied to the multi-redundant Syncom II solar array configuration and used to determine the optimum number of solar cells required to provide sufficient power over the orbital lifetime of the spacecraft. Figure 4-34 presents the results of such an analysis for the Syncom II solar array configuration consisting of 64 groups of cells each with three multi-redundant series-parallel strings with three cells per row and 64 rows per cell group. This figure shows the expected number of series-parallel string groups having r random failures as a function of orbital lifetime.

The derivation of the figure may be illustrated by the following calculations for an orbital time of 5 years and individual cell failure rate λ_c = 0.01 percent per thousand hours. The reliability of an individual cell as expressed by Equation 4-1 is

$$p_{cell} = 0.995618.$$

Since the unreliability is, one minus the reliability becomes

$$q_{cell} = 1 - 0.995618 = 0.004382.$$

The probability of failure of (r) cells, as expressed by Equation 4-3, becomes (r = 0, 1, 2, 3):

$P(0) = p^3 = 0.9869115$ exactly zero failures

$P(1) = 3p^2q = 0.0130310$ exactly one failure

$P(2) = 3pq^2 = 0.0000574$ exactly two failures

$P(3) = q^3 = 0.0000001$ exactly three failures.

The probability that at least one row will have exactly r failures may be written as Equation 4-6.

$$P(r=o) = 1 - [p(r+1) + p(r=2) + p(r=3)] = 0.4306$$

$$P(r=1) = 1 - e^{-(64)(0.013031)} = 0.5657$$

$$P(r=2) = 1 - e^{-(64)(0.0000574)} = 0.0037$$

$$P(r=3) = 1 - e^{-(64)(0.0000001)} = 0.0000$$

The expected number of string groups having r failures from Equation 4-7 becomes (N = 64):

$$E(0) = (64)(0.4306) = 27.56 \approx 27$$

$$E(1) = (64)(0.5657) = 36.20 \approx 36$$

$$E(2) = (64)(0.0037) = 0.24 \approx 1$$

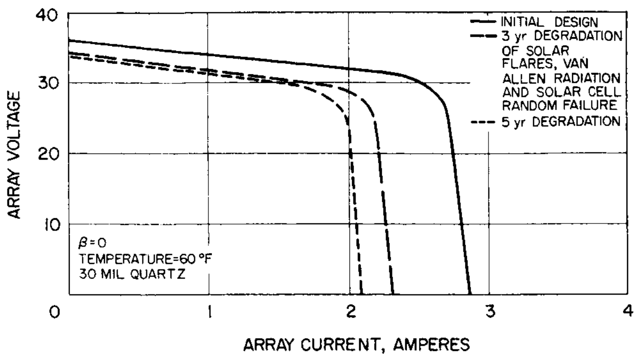
$$E(3) = (64)(0.0000) \approx 0$$

This analysis shows that random failures will be distributed such that the probable occurrence of two or three failures within any row of a string group would be infrequent. Therefore, the solar array output will "gracefully degrade" toward approximately 2/3 maximum power. The loss of any one cell per row in any group being approximately equal to the loss of 1/3 power output of that group.

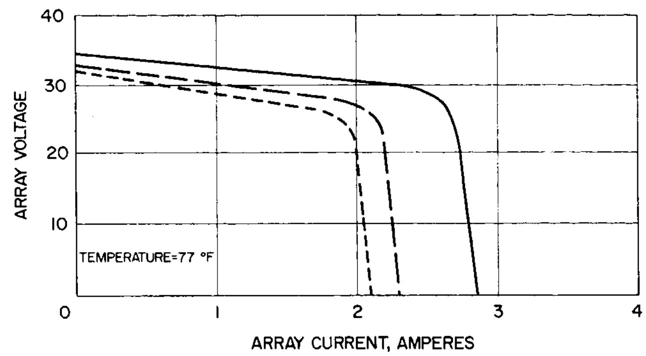
The 64 groups of cells used in this analysis have been selected as representative of one of two Syncom II power supplies. The combination of the results presented for the occurrence of random failures with the effects on solar cell lifetime due to radiation would result in a characteristic plot of the expected solar array output during the orbital lifetime and the optimization of the number (N) of cell groups. This optimization would be based upon the spacecraft power demand during the orbital mission and the expected survival of some spacecraft beyond the average lifetime or mean-time-to-failure.

In addition to the assumption of an independent and random distribution of solar cell failures, a second assumption in applying this model is that the primary failure mode of the solar cell is an open circuit which occurs at random with constant failure rate (λ_c = .01 x 10^{-5} failures per hour). The probability of an individual solar cell failing in a short circuit is negligible, and shorts to ground within the array are assumed not to occur.

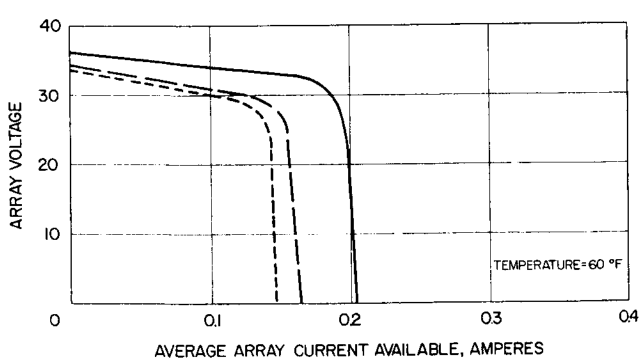
In the Syncom II configuration, each series-parallel string group will be connected to the bus through an isolating diode whose primary failure mode is a short circuit. Thus, the failure of any diode will only increase the reverse current leakage, but will not cause the loss of power from a complete series-parallel string group. For this reason these diodes are not included in the model. The expected diode failure rate is less than 10^{-7} failures per hour, which indicates that the probability that all diodes will survive the 5-year orbital mission time is extremely high and does not significantly affect the model presented.



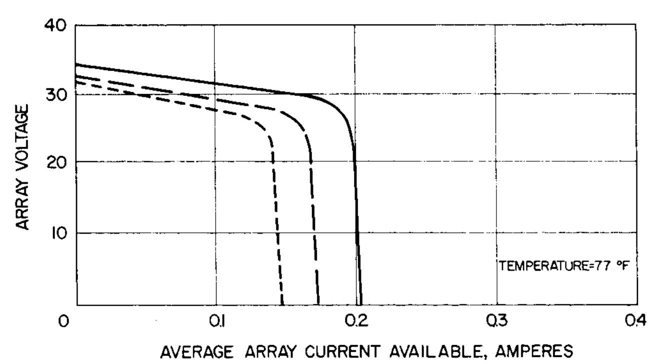
a) Battery array, 60° F



b) Main array, 77° F



c) Main array, 60° F



d) Battery charge array, 77° F

Figure 4-31. Array Characteristics

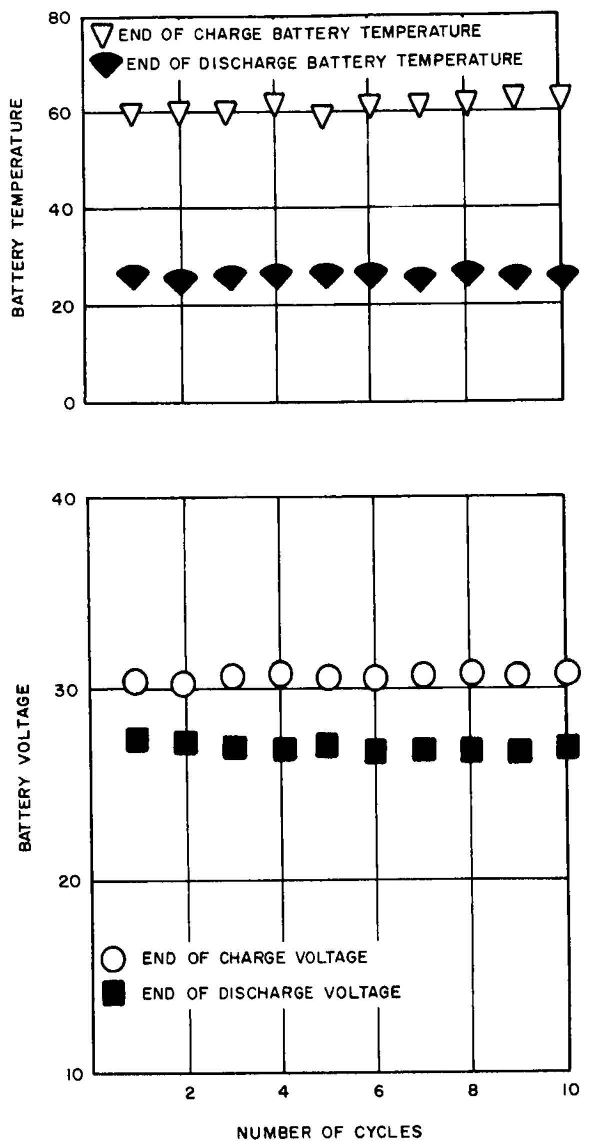


Figure 4-32. Battery Test Simulated

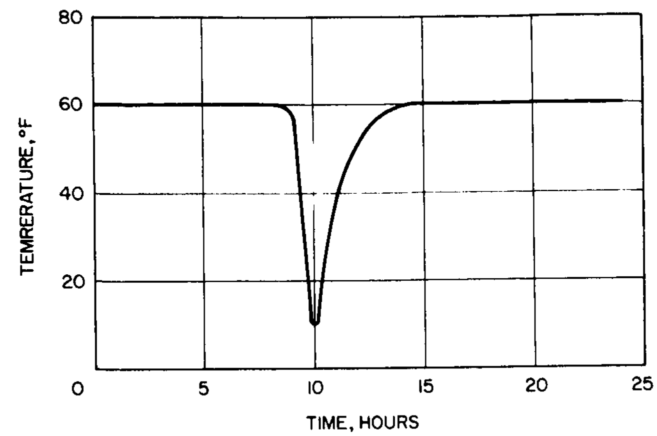


Figure 4-33. Battery Ambient Temperature

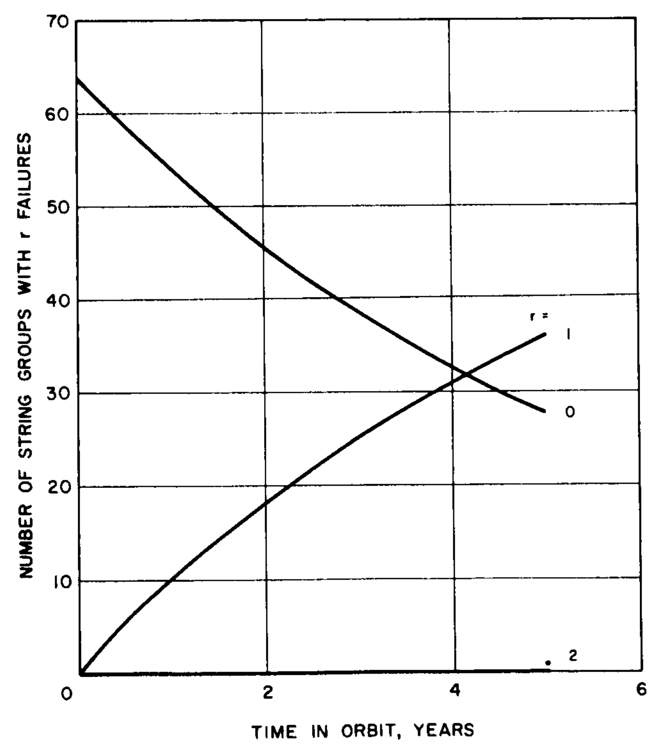


Figure 4-34. Solar Array Degradation due to Random Cell Failures

STRUCTURE

Structural Design

The arrangement of the primary structure is basically unchanged. The engineering effort on structural design during this reporting period has been directed toward combining structure function, toward layout of component support structures to accommodate relocated equipment components and toward the study of advantageous alternate structural arrangements.

The longitudinal stiffening members for the forward tube structure have been located and coordinated to provide continuity to the stiffener material for the total length of the primary structure. The radial bulkheads on the aft thrust tubes are thus connected through the forward thrust tube stiffening members to all the forward truss structure attach points, the forward hoist fittings, and the quadrant electronics support structure.

A study has been completed to determine the optimum structure that connects the forward equipment supporting structure to the thrust tube structure. The comparative evaluation was based on weight, strength, rigidity, and cost. Three structural systems were included in the study: a tubular truss structure, a beaded sheet with large lightening holes, and a vierendeel truss of sheet metal. The tubular truss structure was selected as the optimum arrangement. Due to the extremely light loading, the beaded sheet was rejected because of its excessive weight. The vierendeel truss was possibly 0.10 pound lighter than the tubular truss arrangement but the high shear deflection did not provide sufficient rigidity to this supporting structure. There was no significant difference in cost. Detail design is proceeding on the tubular truss structure.

A conference with Lockheed determined requirements for the structural interface between the spacecraft and the interconnect structure to the Agena vehicle. The alignment shoulder was changed from 90 degrees to 45 degrees contact face to avoid the possibility of tip-off during separation. Also the protruding face was located on the interconnect structure, since this is the stiffer structure and the compressive reaction to the tensile clamping action of the "V" clamp will tend to reduce, rather than increase the clearance gap at the alignment shoulder. The distance between the interface station and the structure outside of the thrust tube was increased to 2 inches. This increase provides greater clearance between spacecraft components and the "V" clamp and is necessary to insure that the "V" clamp during separation may be restrained from striking any spacecraft component or structure. Arrangement of this interface is shown in Figure 4-35.

Structural Analysis

Stress effort during this report period has been concentrated in the following areas:

1) Analysis and design of the three point support solar panels is proceeding. Studies are currently in progress to determine the optimum panel that will meet strength requirements and the deformation limitations of the revised structure.

2) Analysis of the forward equipment support structure is continuing. In addition to the tubular truss structure presently being analyzed, a vierendeel truss and a sheet metal structure will also be investigated.

3) A stiffness comparison of the T-1 and T-2 thrust tube is still in progress.

A 6-degree-of-freedom lumped mass lateral model has been devised for the T-2 engineering model and is shown in Figure 4-36.

Preliminary studies indicate that the fundamental lateral frequency of the T-2 vehicle will be close to the 49 cps first lateral mode frequency observed in the T-1 vehicle.

Mass Properties Analysis

Structural redesign of the spacecraft accounted for a major portion of the effort since the last reporting period. This resulted in a reduction in the structure subsystem weight of 40 pounds. A comprehensive updating of electronics subsystem weights, however, reflects an increase of 48 pounds. In addition, a required increase in solar cell glass cover thickness provides an increase of 15 pounds in power supply subsystem weights.

Although the current spacecraft weight does not exceed the specified maximum of 1518 pounds, the changes which occurred reduce the contingency factor (ballast subsystem weight) approximately 20 pounds.

Coincident with changes in the mass distribution, the ratios of the roll-to-pitch moments of inertia have effectively been reduced below the specified value of 1.2. Currently studies are being made to determine the optimum placement of components to achieve increases in the roll-to-pitch ratio without compromising weight and other limiting parameters.

Table 4-18 illustrates in detail the mass properties distribution for the Advanced Syncom Model HSX-302-T2. The $\Delta\omega$ values shown are changes in weight from the flight spacecraft since the last report and reflect the major redesign which has occurred. Future changes will be listed, using the present configuration, HSX-302-T2, as a basic model.

To implement weight control, target weights have been established for the respective subsystems. Target weights of vendor items are listed in the respective specifications. Hoist fittings which are non-functional in spaceflight have been eliminated from the spacecraft weight.

TABLE 4-18. ESTIMATED WEIGHT STATUS, HSX 302-T2

Subsystem	$\Delta\omega$ Pounds*	Weight, Pounds	ϕ**	ϕ***	Target Weight pounds
Electronics	+47.7	182.4	0.292	0.120	140.45
Wireharness		19.9	0.032	0.013	16.40
Power supply	+14.0	127.5	0.204	0.084	122.68
Control	- 1.6	47.3	0.076	0.031	48.86
Propulsion		122.2	0.195	0.081	122.80
Structure	-40.0	98.3	0.157	0.065	96.25
Miscellaneous		19.9	0.032	0.013	16.40
Ballast	-19.8	8.2	0.012	0.019	62.50

	Weight, Pounds	Z-Z, Inches	I_{z-z}, Slug-ft^2	I_{x-x}, Slug-ft^2	R/P
Final orbit condition	(625.7)	21.5	57.0	53.1	1.07
N_2 pressurization	2.9				
N_2H_3-CH_3 fuel	53.0				
N_2O_4 oxidizer	84.1				
Total at apogee burnout	(765.7)	21.5	71.3	60.3	1.18
Apogee motor propellant	752.3				
Total payload at separation	(1518.0)	22.6	88.1	77.5	1.14

* Change in subsystem weight since last report
** Ratio of subsystem weight to final orbit condition weight
*** Ratio of subsystem weight to total payload at separation

Dynamic Test Status and Preliminary Data Analysis (T-1)

Model analyses of the T-1 structure are essentially complete and show good agreement with the results of the T-1 vibration test. The calculated longitudinal fundamental frequency is within 10 percent of the measured frequency (125 cps), and the calculated lateral fundamental frequency is within 4 percent of the measured frequency (49 cps). Comparisons of the fundamental mode shapes will be completed after measured responses of the apogee motor propellant are reduced from taped data. Preliminary comparisons of these modes show satisfactory agreement. The results of the model analyses will be published in the near future.

Sinusoidal vibration testing of the T-1 structure has been completed and the data are being reduced to provide design loads and component test environments for the new spacecraft configuration. Vibration response envelopes have been submitted to Lockheed/Sunnyvale for use in their dynamic studies (Figures 4-37 and 4-38).

Wiring Harness Design

The interconnecting wiring harness will be made of only those materials which have been tested and found suitable for use in space. Effort is being made to maintain the harness weight to a minimum through the use of light weight wire, terminal boards, connectors and other hardware. Investigations into improved methods of fabrication and assembly of the harness are in progress. One method under consideration is the use of new butt welding techniques. Vendors have also been contacted in regard to improved components. However, the basic concepts of maximum reliability, minimum sublimation, weight, and size without sacrificing quality will determine the components used.

The configuration of the harness assembly is determined by the spacecraft structure. The basic design under development provides for a harness composed of two separate assemblies, each consisting of a semicircular main section with breakouts terminating at the electronic unit packages, telemetry, power, control, communications, and propulsion equipments. These two assemblies may be joined together with mating connectors to form a complete unit, if indicated by system design; this requirement is under study at the present time.

This use of two separate assemblies, a deviation from the design for Syncom I, will result in a reduction of weight and materials and permit installation of the main harness section in the aft area of the spacecraft which is preferred to meet weight and balance design objectives.

The harness will be installed through and secured to the 12 support ribs spaced around the circumference of the aft section. Previously the harness was to be located outboard of the forward bulkhead in the forward section of the spacecraft. Installation of the harness aft will provide increased protection of the harness from physical damage and will eliminate the possibility of interference between the harness assembly and the electronic quadrant packages or the propulsion system tanks and plumbing.

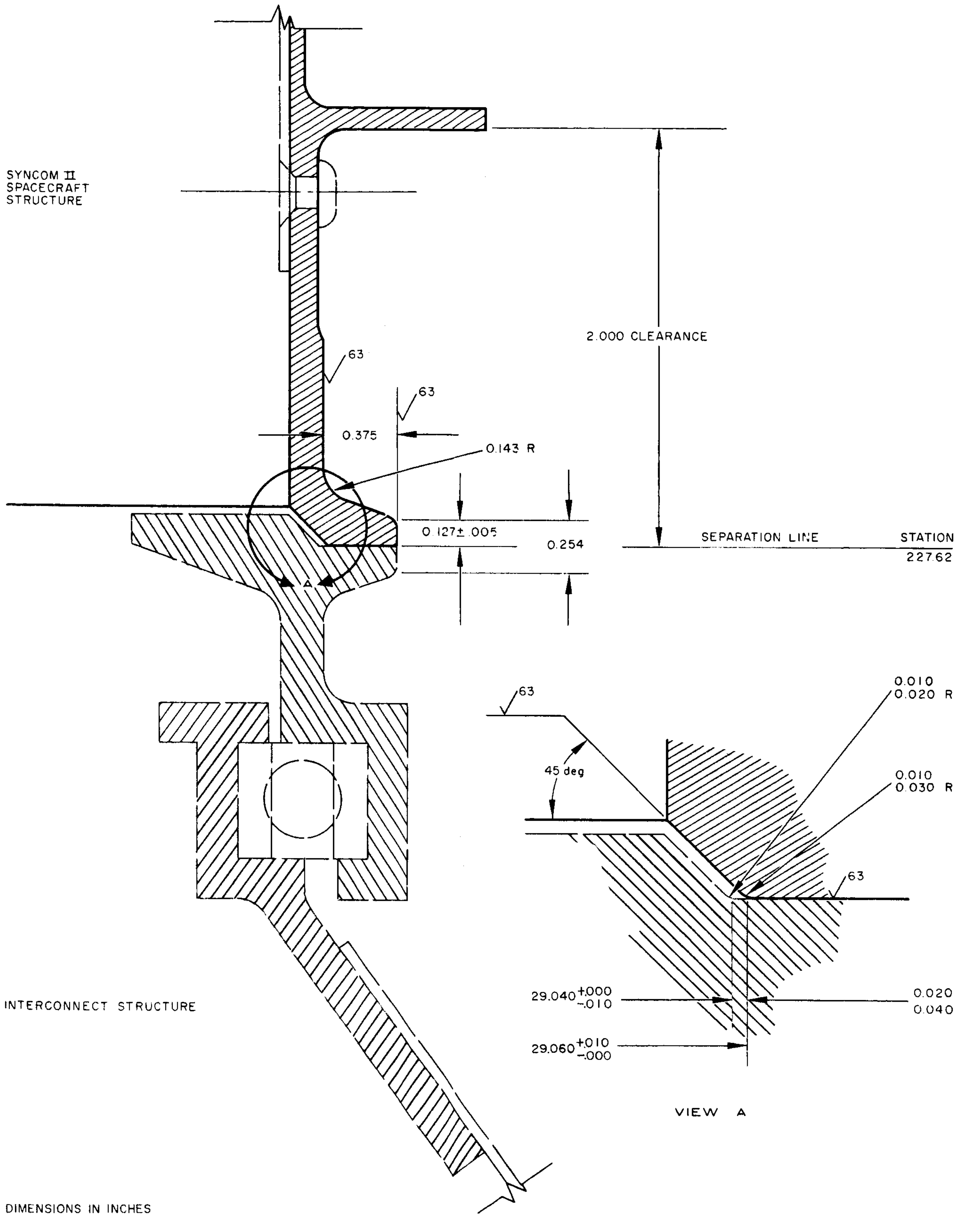


Figure 4-35. Ring, Thrust Tube

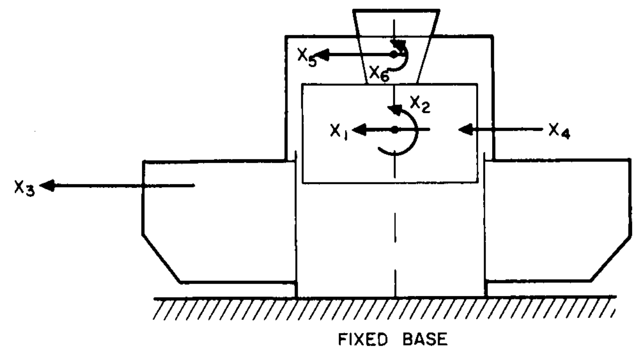


Figure 4-36. Lateral Model Showing Mass Stations and Coordinates

Mass Station	Weight
1	Mass of apogee motor plus propellant
2	Mass moment of inertia of the apogee motor
3	Mass of ribs, thrust tube, and half the mass of bipropellant tanks
4	Mass of half the bipropellant tanks, quadrant electronics, stiffeners, and center tube
5	Mass of the forward structure
6	Mass moment of inertia of the forward structure

Coordinates 2 and 6 are rotational. All of the others are translational.

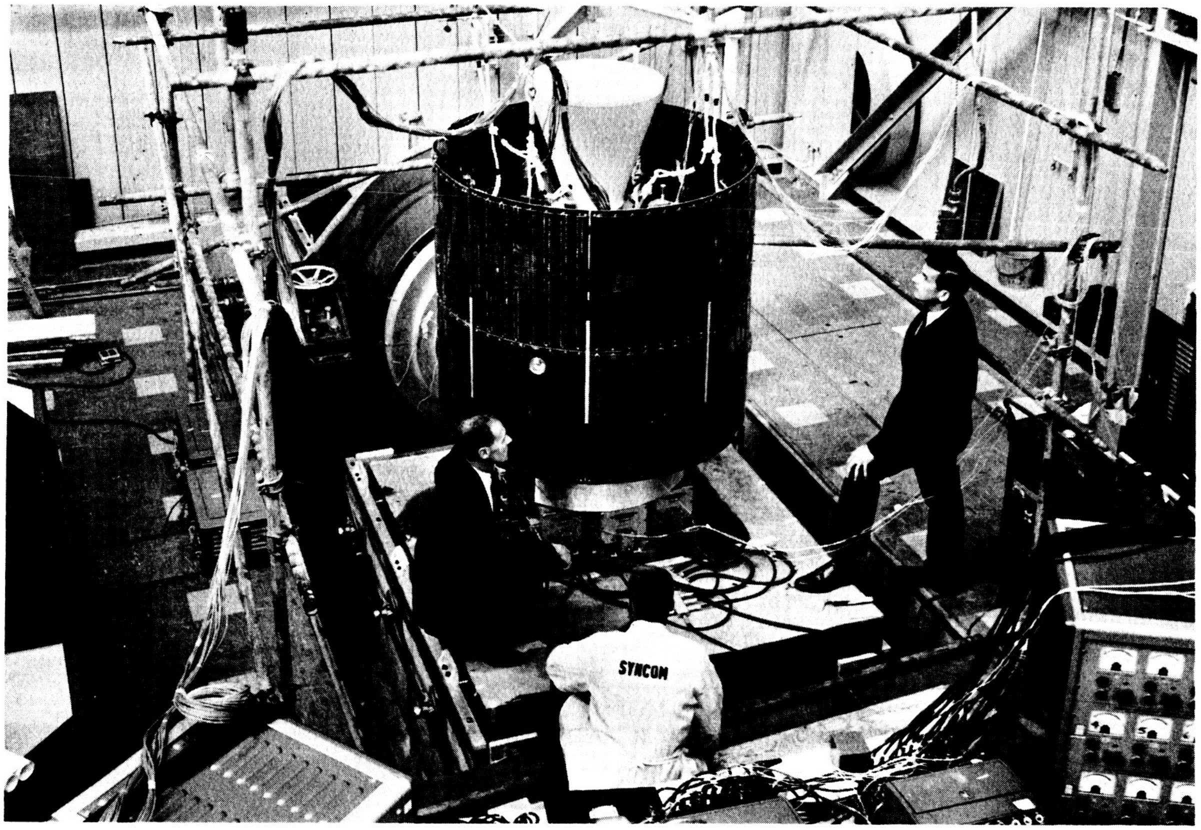
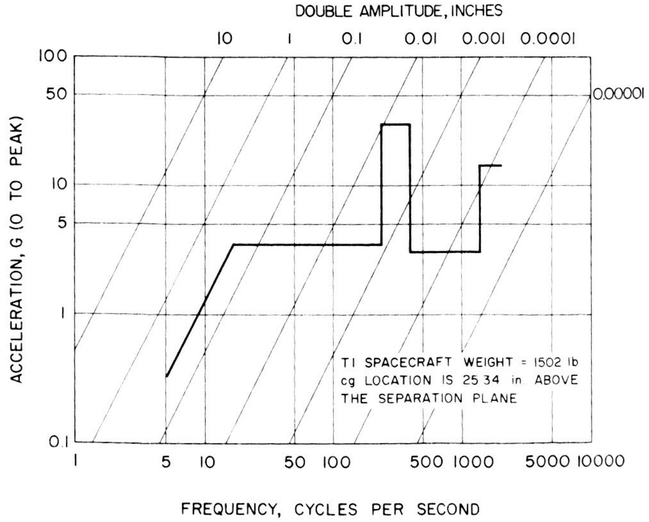
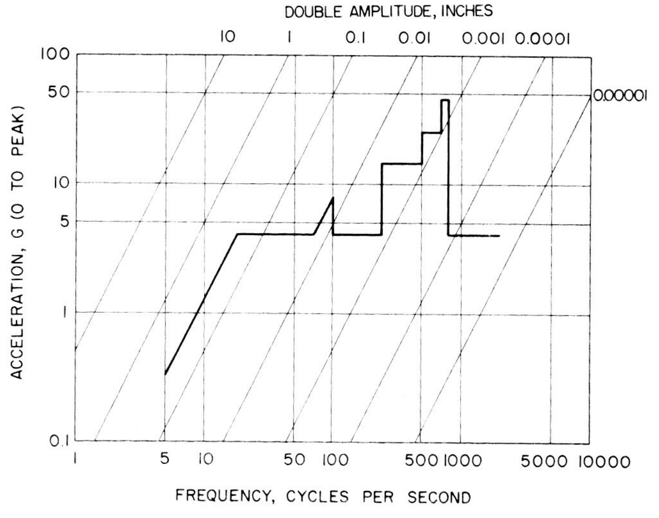


Figure 4-37. General View of Vibration Test Setup



a) Lateral



b) Longitudinal

Figure 4-38. Vibration Response Envelopes

The preceding design and installation concepts are predicated upon resolution of certain details involving relocation of electronic package connectors, pending relocation of some battery units and TWTs. These revisions are expected to improve pitch and yaw characteristics of the spacecraft.

Nutation Damper Design

The preliminary design for determining the Syncom II nutation damper dimensions has been completed and a test program is being formulated for evaluating the effectiveness of the design. The changing of the kinetic energy of nutation to heat is performed by the dampers in an acceleration field greater than 1 g. However, the tests will be made on a dynamically similar model in a 1-g field. The dimensions of this model have been determined by requiring that certain force ratios for the model and the actual damper be made equal. The forces used for the development of these ratios are:*

1)	Driving force	$\alpha\ \rho\ \omega^2\ dL^3$
2)	Viscous force	$\alpha\ \eta\omega L^2$
3)	Acceleration force	$\alpha\ \rho\ aL^3$
4)	Inertia force	$\alpha\ \rho\ \omega^2\ L^4$

For similarity, these force expressions have the same ratios in the model and the actual dampers and yield the following expressions:

$$\omega_2 = \left(\frac{L_1}{L_2}\right)^{1/2} \left(\frac{a_2}{a_1}\right)^{1/2} \omega_1$$

$$d_2 = \left(\frac{L_2}{L_1}\right) d_1$$

and

$$v_2 = \left(\frac{a_2}{a_1}\right)^{1/2} \left(\frac{L_2}{L_1}\right)^{3/2} v_1$$

where subscript 2 refers to the model damper
Subscript 1 refers to the actual damper

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* D. D. Williams, "The Undamping Effect of Hydrogen Peroxide Sloshing in Syncom," IDC, 29 November 1961.

and L = length of damper tube

ω = angular frequency

a = acceleration

d = diameter of damper tube

v = kinematic viscosity

These scaling relationships result in the model damper and the actual damper having the same Reynolds number (inertia force/viscous force) and Froude number (inertia force/acceleration force). By scaling the energy dissipation rates of the model and the actual system, the time constant for the prototype damper can be determined by tests with the model on a pendulum of appropriate length (i.e., the length of a simple pendulum corresponding to ω_2). The fabrication of the test equipment and the development of the test procedures will begin during the next report period.

THERMAL CONTROL

Thermal Nodal Analysis

The preparation of the complex thermal network representation of the Syncom II vehicle is continuing. Structure, tankage, electronics, and associated tasks are being included in a lumped parameter system representative of one-fourth of the Syncom II spacecraft. Requests have been made for thermal information inputs from every control item on the vehicle. These inputs include maximum and minimum allowable component temperatures, desired orbital temperature, and power dissipation in the low power, high power, and expected orbital power modes.

Vehicle Parametric Analysis

A preliminary bulk thermal analysis of Syncom II has been completed in an attempt to establish the significance of various parameters in their effects on the basic thermal design. In order that all significant parameters be investigated, a simplified analytical model of the vehicle was utilized in which several compromises were made. The actual Syncom II vehicle consists of basically five external surfaces; i.e., solar panels, ground plane, thermal barrier, apogee motor nozzle, and phased array antenna. These surfaces, in addition to the internal power dissipation of the vehicle, determine the vehicle bulk temperatures in the deep space environment. The parametric survey was made based on a simplified Syncom II model, wherein the phased array antenna and the apogee motor nozzle were neglected in the bulk thermal analysis. The model assumed a right circular cylindrical geometry consisting of solar panels, thermal barrier, and ground plane. Parameters that were varied were:

1) Solar angle

2) Internal power

3) End planes surface properties

4) Solar panel properties

5) Earth orbital variations

The thermal analysis was also extended to include an error type analysis to evaluate the degree of uncertainty in the detailed temperature predictability of a Syncom II type of vehicle. In addition to basic uncertainties in the above mentioned parameters, other factors contributing to the overall tolerance on a specific temperature are:

1) Power dissipation (at a given point)

2) Thermal conductance uncertainties

3) Thermal radiation coupling uncertainties

The variations of the parameters were applied to a vehicle design nominally resulting in an orbital equilibrium temperature of 70°F. The nominal parameters that resulted in this equilibrium temperature were:

1)	Solar panel emittance	$\bar{\epsilon}$ = 0.82
2)	Solar panel absorptance	$\bar{\alpha}$ = 0.85
3)	Thermal barrier surface	vapor deposited aluminum
4)	Ground plane surface	40 percent white paint; 60 percent vapor deposited aluminum
5)	Solar angle	γ = 90°F
6)	Internal power	140 watts

The changes in vehicle bulk temperature that occurred for given perturbations in the parameters are shown in Figure 4-39. The vehicle bulk temperature is seen to be extremely sensitive to such things as solar panel absorptance, solar panel emittance, and earth orbital position in respect to the sun, but was only slightly affected by ground plane emittance and thermal barrier emittance.

All of the parameters investigated along with known or expected reasonable perturbations to the nominal values are listed in Table 4-19. The corresponding fluctuation in vehicle bulk temperature is shown for each parameter. Additional possible perturbations or errors that are applied to an internal nodal temperature calculation are listed in Table 4-20. They are:

1) Nodal power variations

2) Modeling inaccuracies

   a) Conductance simulation

   b) Radiation simulation

The root sum square deviation to a calculated nominal temperature is calculated in Table 4-19. On the basis of a bulk temperature estimate for a Syncom II type of vehicle, a tolerance of ±11°F must be applied. If the additional inaccuracies of the thermal analytical model of the internal nodal points of the vehicle are included, it is seen that the nodal temperature tolerance is ±14°F. In the actual case however, several parameters can be relatively fixed. These are:

1) Spacecraft power dissipation (within 5 percent)

2) Solar angle on vehicle (time of year)

3) Earth orbital position

The root sum square calculation is shown in Table 4-20 and is seen to result in a tolerance of ±12°F when applied to a nodal temperature calculation. Conclusions derived from this analysis are:

1) The thermal barrier surface must be present on the vehicle for satisfactory thermal control. If no barrier exists, the vehicle temperature cannot be raised above 55°F when the sun is on the opposite end of the spacecraft from the apogee motor.

2) A tolerance must be assumed of ±12°F on any computed nodal temperature even after best engineering practice has been applied both to analytical methods and surface property measurement.

Reaction System Analysis

Preliminary thermal analysis of the Syncom II axial reaction control nozzles has indicated maximum and minimum steady state non-operating temperatures of 0° and 90°F respectively. Calculations have been based on steady state orbit temperature conditions, low transmitter power with sun angle of 115 degrees and high transmitter power with sun angle of 65 degrees. Solar absorptances, infrared emittance, and boundary temperatures where applicable, have been taken from Syncom I data.

TABLE 4-19. SYNCOM II TEMPERATURE RESPONSE TO PARAMETRIC VARIATIONS

Parameter	Possible Deviation from Nominal	$+\Delta T$, °F	$-\Delta T$, °F	$+\Delta T^2$	$-\Delta T^2$
Bulk					
Solar angle	±25 degrees	2	0	4	0
Internal power	±5 percent	2	2	4	4
Solar absorptance					
1) Ground plane	±5 percent	2	2	4	4
2) Thermal barrier	±5 percent	2	2	4	4
3) Solar panel	±5 percent	6	6	36	36
Emittance					
1) Ground plane	±5 percent	1	1	1	1
2) Thermal barrier	±5 percent	0	0	0	0
3) Solar panel	±5 percent	5	5	25	25
Orbital variations	±4.4 percent	7	7	49	49
Sub total		+27	-25	127	123
Nodal					
Power	±5 percent	1	1	1	1
Conductance	±20 percent	8	8	64	64
Radiation coupling	±20 percent	3	3	9	9
Total		39	37	201	197

Bulk temperature variation (rss): $T = T_{nominal} \begin{smallmatrix} + \sqrt{127} \\ - \sqrt{123} \end{smallmatrix} = T_{nominal} \pm 11°F$

Nodal temperature variation (rss): $T = T_{nominal} \begin{smallmatrix} + \sqrt{201} \\ - \sqrt{196} \end{smallmatrix} = T_{nominal} \pm 14°F$

The thermal analysis included:

1) Radiation coupling between reaction control nozzle and all adjacent surfaces.

2) Direct solar irradiation

3) Reflected solar energy

Conduction coupling was exluded from the analysis since the manner of attachment, and the extent of the structure is at this time relatively unknown. It is expected that an increase of 15 to 30°F to the lower temperatures is achievable with a reasonable amount of attachment conductance to basic structure.

Solar Panel Thermal Analysis

Calculations have been performed on the Syncom II solar panel to determine the range of expected temperatures. The desire has been stated to reduce the temperature level of the panel to increase panel electrical output. Since the orbital temperature level of the Syncom II solar panel is basically a function of the solar absorptance of the panels, IR emittance of the panels, and the degree of thermal coupling to the Syncom II vehicle, investigations have been made to determine the effect of varying the surface properties. The solar cell properties themselves are of course the major factor in determining the panel temperature, but the effect of the surface properties of the material between and around the individual cells is definitely not negligible.

The surface substrata for the Syncom I solar panels is a black paint, PT-401, of the surface properties α solar = 0.92 and $\epsilon_{\overline{IR}}$ = 0.85. If an inorganic white paint such as the Hughes white is used for the Syncom II solar panel painted regions (which account for 10 percent of the total surface area), the temperature of the Syncom II solar panel can be reduced nominally by 12°F based on the surface properties of the white paint of α solar = 0.18 and $\epsilon_{\overline{IR}}$ = 0.85.

Preliminary estimates as to the eclipse cooldown rates for the solar panel and battery on Syncom II have been made. Figure 4-40 indicates maximum expected cooling rates based on a 70 minute eclipse and previously calculated Syncom I eclipse transient temperatures.

Space Effects on Thermal Control

Although the space effects on thermal properties of proposed Syncom II outside surfaces are not known, an attempt has been made to evaluate vehicle sensitivity to these possible effects. A simplified analytical model of the Syncom II spacecraft has been analyzed as to the effect on bulk temperature of changes in the surface nominal thermal properties that may occur due to ultra violet exposure, micrometeorite impingement, apogee motor blast, etc.

The analytical model was parametrically evaluated for changes in solar absorptance and IR emittance of the thermal barrier, ground plane, and solar panels. Results for various degrees of degradation are shown in Table 4-21. It is seen for example that a 5 percent change in all properties results in a root sum square (rss) temperature change of 8.4°F. Other degrees of property degradation are also shown If the degradation is stated on a per year basis, the rate of bulk temperature change may be seen in Figure 4-41 to vary significantly over the life of the spacecraft. For instance, at the rate of 5 percent per year degradation to all coatings, the spacecraft temperatures can be expected to vary from initial temperatures by approximately 25°F at the end of a 3 year period.

It is emphasized that the surface degradation rates are unknown. However, it is felt that a reasonable rate to expect would be in the 5 to 10 percent per year range at least for the initial months of the spacecraft life and a gradual tapering off to lower rates as the vehicle mission progresses. As the actual surface exposure test data becomes available, the expected vehicle temperature excursions can be incorporated into the thermal design.

APOGEE INJECTION ROCKET ENGINE

The apogee motor is under development by JPL, and is to be a 2.33:1 scale-up of the Syncom I apogee motor All components with the exception of the flight nozzle have been designed. To fully evaluate candidate materials and fabrication techniques, a sub-scale nozzle program, utilizing Syncom I apogee motor hardware, has been initiated. Two completely successful tests have been conducted to date to determine the relative merits of modified versus basic phenolic as an impregnating material. The fired components have been sectioned, and it appears that glass fiber impregnated with modified phenolic is less subject to charring. At the present time parallel design effort is being exerted on both conical and contour nozzle expansion sections. The optimum design will be selected on the basis of weight/ performance trade-offs.

Two tests of a series designed to optimize the igniter configuration (as a trade-off between pyrotechnic and controlled-pressure-types) have been conducted. Both involved the use of an ignition test motor which duplicates the chamber volume and throat area of the apogee motor. As data reduction is not complete, no results are available at this time. The first heavy wall test will be conducted in July, and the first flightweight test in October, 1963. Flight motor delivery will be initiated in September, 1964.

TABLE 4-20. SYNCOM II VARIATIONAL TEMPERATURE TOLERANCES

Parameter	$\pm \Delta T^2$	$- \Delta T^2$
Power dissipation	5	5
Solar absorptance	44	44
Emittance	26	26
Modeling inaccuracies	73	73
Total	148	148
Nodal temperature	$T_{nominal} \pm 148$	
$T = T_{nominal} \pm 12°F$		

TABLE 4-21. SYNCOM SPACECRAFT SENSITIVITY TO SURFACE DEGRADATION

Surface Property Change	$\pm \Delta T°F$			ΔT^2		
	2 1/2%	5%	10%	2 1/2%	5%	10%
Solar cell absorptance	3	6	12	9	36	144
Solar cell emittance	2.5	5	10	6.25	25	100
Ground plane absorptance	1.0	2	4	1.0	4	16
Ground plane emittance	0.5	1	2	0 25	1	1
Thermal barrier absorptance	1.0	2	4	1.0	4	16
Thermal barrier emittance	0	0	0	0	0	0
Total	8.0	16	30	17.5	70	277
RSS, °F				±4.18	±8.4	±16.6

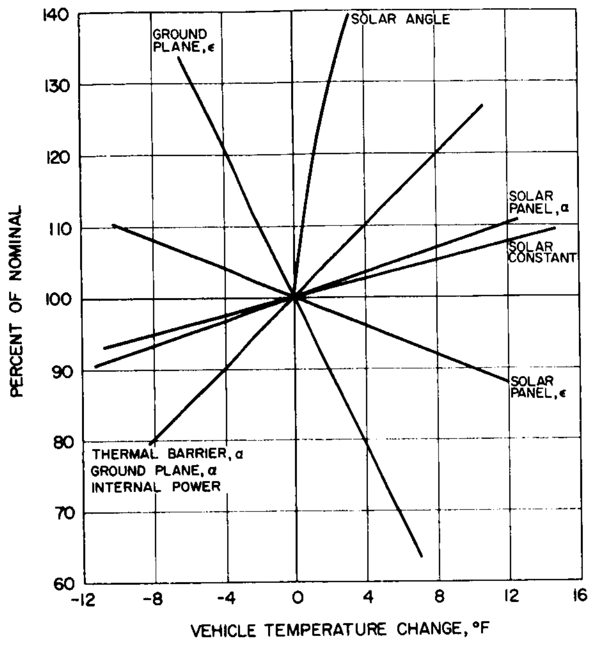


Figure 4-39. Thermal Effects of Vehicle Parameters

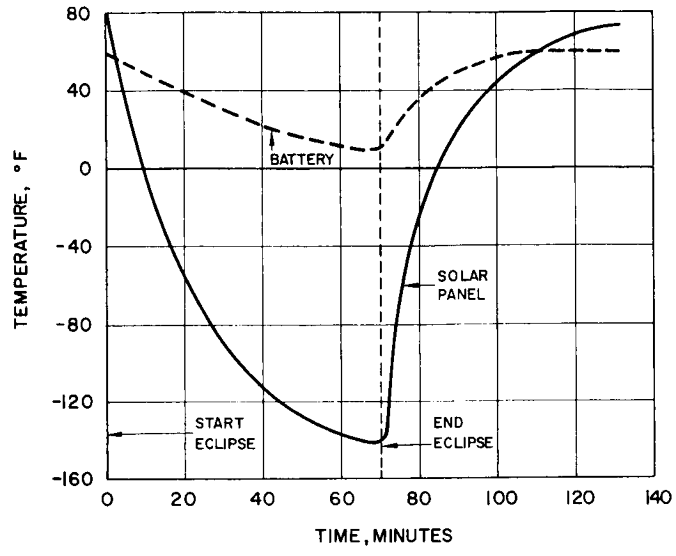


Figure 4-40. Power Systems Eclipse Temperatures

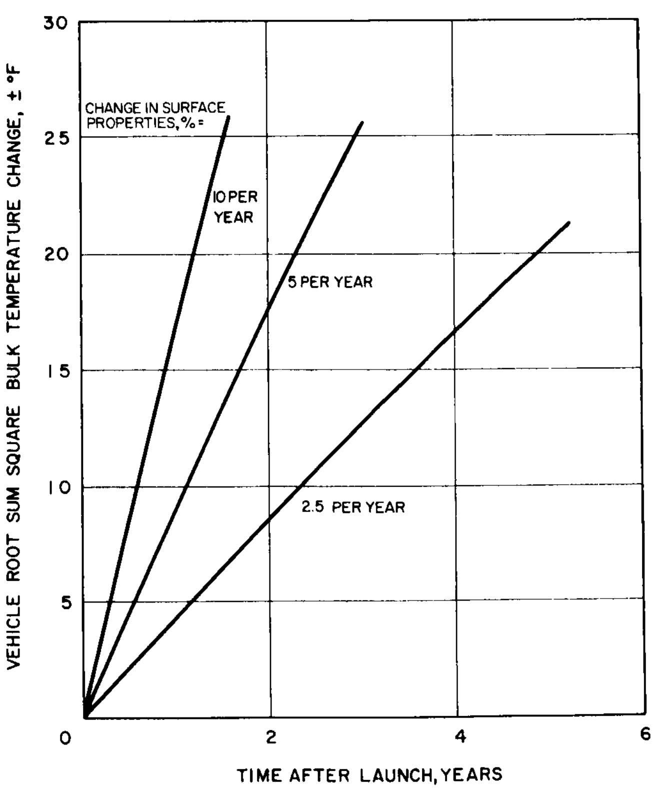


Figure 4-41. Durface Degradation Effects on Vehicle Temperatures

5. SPACECRAFT RELIABILITY AND QUALITY ASSURANCE

QUALITY ASSURANCE

The Quality Control Operating Plan was finalized during this report period. Applicable quality control documentation requirements have been revised to reflect the contractual requirements for this phase of the program. The requirements established for Level I Quality will apply since only engineering models will be fabricated during this time. In-process inspection is being performed on items for these engineering models. Corrective action requests were completed on facilities with a "conditional" or "disapproved" rating and were forwarded to the respective vendors.

A re-survey of four semiconductor facilities will be required to verify corrective action for compliance with NASA NPC 200-3. Quality Control monitoring will be required on 15 facilities. The surveys on all passive device suppliers have been completed. A re-survey of nine facilities will be required to verify compliance to NASA NPC 200-3. Quality control monitoring will be required on 11 facilities.

The Supplier Control Group has initiated a Quality Plan for passive device suppliers.

The Marquardt Corporation was surveyed for compliance to NASA NPC 200-2 and Hughes requirements. There were two major discrepancies: inspection/test equipment were overdue for calibration, and a quality control plan had not been established. Corrective action will be required.

STATUS OF INSPECTION INSTRUCTIONS

Quality Assurance has reviewed the Syncom Mark II requirements for Inspection Instructions. In fulfillment of this requirement certain actions have been taken: selection of format, interpretation of the specified requirements (Section 7.3.1 of NPC 200-2), and initiation of inspection instructions to presently available drawings. Inspection instructions will be issued for all major and minor control items, the subassemblies involved, and certain critical parts.

It is anticipated that it will be periodically necessary to prepare a series of inspection instructions to meet the exigencies of fabrication and assure complete and adequate coverage. To ensure an effective operation, efforts will be made to coordinate the various inspection processes and the associated instructions with laboratory planning functions. The Quality Control Inspection Planning Sheet, presently in use, will be used in implementing the inspection requirements of NPC 200-2. This form is considered adequate, but specific coverage of the following points in each procedure is necessary to assure compliance with NPC 200-2.

1) The objectives of the inspection should be stated in all cases. As indicated in the proposed constraints, no waiver will be required.

2) The requirement for providing a listing of the measuring equipment to be used and details of the operation to be performed (scales to be read and the exact method of inspection) should be waivered. These requirements are excessive when they involve the use of ordinary vernier calipers, micrometers, and similar items which are completely familiar to skilled and experienced inspection personnel. However, detailed instructions will be used in the case of unusual layouts or complex items requiring special measuring instruments. These instructions will be implemented when necessary.

3) Conditions to be maintained during inspection and necessary handling precautions will be covered by a general instruction; in special cases they will be defined in the inspection instruction.

4) Acceptance and rejection criteria including the tolerance allowed will be specified for each point of inspection. Where listing of such criteria becomes excessive, the listings will be limited to the critical parameters.

Inspection instruction sheets are presently being prepared, using those available drawings on Spacecraft T-1 which are not expected to undergo extensive changes. As the new drawings are released, these inspection instructions will be revised. The spacecraft structure drawings are presently undergoing extensive changes; consequently, these instructions will not be prepared until firm or near firm drawings are available. Inspection instructions have been prepared for etched boards and welded modules. These instructions were prepared from similar Syncom I released drawings and Syncom II pre-release drawings.

6. MATERIAL, PROCESSES, AND COMPONENTS

CRITICAL COMPONENT TESTING

Preferred Parts List

The preferred parts, material, and process list of the April 1963 Supplement to the Summary Report (Appendix) has not been changed during the reporting period. Qualification tests are proceeding on 16 semiconductors which will be added to the list when completed, and several units will be deleted in the next revision of the list.

The status of critical component testing is as follows:

1) Ferrite switches. The switches are presently in the breadboard stage and tests have not yet been initiated. It is expected that the switches will be ready for test by the end of July.

2) Phase shifters. Temperature tests have been initiated on three phase shifters; however, no results have been obtained to date. It is expected that during the next period this testing will be approximately 90 percent complete.

Bipropellant Injector Solenoid Valve Assemblies

The solenoid valve tests have been revised to include the following tests during Phase I:

1) Two valves shall be subjected to repeated temperature cycling. Each valve will be pulsed a minimum of 25,000 times during temperature cycling using actual flight propellants. Before and after each temperature and environment cycle, the units will be tested for pulse characteristics and for leakage. The actual solenoid drivers will be used to excite the valves for all operational tests. Transient effects and power input will be measured as a function of temperature.

   After temperature cycling tests, the valves will be subjected to qualification level vibration environment. No propellants will be used during vibration tests.

2) Two valves will be endurance tested for a minimum of 1 million pulses each. The units will be tested for pulse characteristics and for leakage at predetermined intervals and at the conclusion of all tests.

The testing for the valves has not yet begun, but valves are being ordered and the solenoid drivers are being fabricated.

7. SPACECRAFT SUPPORT EQUIPMENT

INTERFACE SPECIFICATION

Agena D/Spacecraft Interface

NASA has issued a followon contract to Lockheed to perform a design study of Agena D/Spacecraft interface. The study will result in a preliminary design of the spacecraft/Agena D interface ring, spin table, rocket spin up system, RF transparent adapter section, and nose shroud.

A series of NASA, Hughes and Lockheed coordination meetings have been conducted. The meetings have served to define Agena D/Spacecraft interfaces, identify interface problems that require further study, and establish working relationships between the three organizations. Interface problems requiring further resolution include spacecraft clearance under the Nimbus shroud and spin table acceleration characteristics.

GROUND SUPPORT EQUIPMENT SYSTEM BLOCK DIAGRAM

The system block diagram (Figure 7-1) represents the test equipment that will perform all electronic spacecraft system tests. The configuration shown permits semi-automated test procedures. The diagram indicates the major equipment classification areas. Details of each area can be found on subsequent pages as the communication subsystem, telemetry and command subsystem, recording subsystem, and frequency measurement and time standard subsystem (Figures 7-2 through 7-5).

Semiautomation is accomplished through the use of programmed patch boards. The control console has provisions for identification of the patch board in use. In addition to monitoring critical spacecraft functions, one or more tests may be carried out with a preprogrammed recording arrangement that will allow quick-look, detailed analysis and permanent record storage. Commercial equipment that will permit communication system tests on the spacecraft is included and will be capable of carrying out tests such as envelope delay and linearity, noise loading, TV signal distortion, and similar experiments.

GROUND SUPPORT EQUIPMENT SPECIFICATIONS TREE

Ground support equipment is to be provided for maintaining and verifying the proper operation of all spacecraft subsystems, calibration of all sensors and telemetry systems, and monitoring and control of spacecraft functions during system tests and launch operations.

The Specifications Tree (Figure 7-6) indicates specifications that are to be written and how they interrelate from a system point of view. The specifications will define both the equipment and necessary tests to ensure the proper operation of this equipment.

COMMAND GENERATOR

The command generator console consists of the three panels shown in Figures 7-7a, 7-7b, and 7-7c. The left panel contains the dc power panel, the oscilloscope for displaying wave shapes to the test control operator, the intercom control, and the display panel showing various displays required for test control. The dc Power Supply panel on the console contains all controls to select and monitor the appropriate power supply. Switches enable the operator to select the mode of operation, either spacecraft power (internal batteries) or exterum power (dc power supply). The power supplies are monitored by voltmeters and ammeters. The voltmeters will measure the voltage at the power supply or at the test fixture when the connector power switch is de-energized or energized, respectively. The ammeters will measure either the load current or the charging current depending on ammeter selection. The connector power switch is a safety feature which must be energized to apply power from the dc power supply panel to the test fixture. Two Harrison 810B power supplies will furnish the required power for all spacecraft operations controlled through the panel.

The central panel contains the controls for selecting, transmitting, and executing commands, ground controls for antenna position and jet firing, indicators for displaying the status of the spacecraft electronics subsystems, and controls to perform a self test of the command generator electronics. The right panel (telemetry display panel) contains selectable digital and analog readouts, miscellaneous status displays, and controls for selecting the telemetry system to be displayed.

Preliminary logic design to implement the command console is 85 percent complete. This figure includes logic for primary and secondary modes of operation, separate transmission of address and command, octal display of command information, and verify-error detection logic. Internal test logic for testing octal displays and the logic related to primary and secondary modes of operation has been completed.

Logic is in the process of being written for command legend display, spacecraft status display, elapsed time indicators for JCE, execute mode selector, and interlock commands. Logic for execute angle selection has been completed. This portion of the synchronous controller is being integrated into the command generator execute mode selector.

Telemetry Processor

A possible technique for reconstructing the pulse amplitude modulation (PAM) telemetry waveform from the telemetry video is being examined for its feasibility. This technique utilizes combinations of bandpass filters, frequency discriminators, bias voltage supplies, and switching circuits. The approach is desirable because the testing of Syncom II is most effective when the PAM waveform is used.

Figure 7-8 is a preliminary block diagram of the telemetry processor. The telemetry format is the NASA standard PFM system, with one reference frequency that is greater than the highest data frequency. The differential amplifier indicates the presence of the reference burst. At the end of the reference burst, the pulse width detector is reset; if the frame sync pulse is present, the pulse width detector switches. The frame sync frequency on odd frames is 4500 cps; the frame sync frequencies for even frames are the discrete frequencies used for telemetering digital information. The sync frequency filter is tuned to 4500 cps; the outputs of the pulse width detector and the sync frequency filter indicate the presence of a sync pulse and identify it as odd or even.

The PAM waveform is gated through electronic switches to the analog-to-digital converter. On even frames, the appropriate bits of the converter output are transferred to the frame counter. On odd frames, the frame counter is stepped by the sync frequency filter. Each reference burst causes the channel counter to step. Each sync pulse causes the channel counter to be reset to zero.

The frame and channel counters drive the various electronic switches for signal routing. External input signals, in the form of dc voltages, can be selected for display on the analog and digital displays by means of control switches (not shown) which permit selection of the inputs applied to the sample-and-hold circuits and to the analog-to-digital converter.

Information displayed digitally is scaled by means of adjustable scale factors (potentiometers) and electronic switches. The scaling is automatically accomplished by switching the appropriate reference voltage to the A to D converter such that the output is numerically equal to the engineering units of the corresponding channel.

Major and Minor Control Item Test Equipment Requirements

To meet checkout schedules when production of Syncom satellites is initiated, the following test sets will be necessary:

1) Automatic Module Tester. This will be used to test most of the spacecraft welded modules and the logic cards for the ground control equipment. Modules will be tested on a go, no-go basis. Those modules found to be defective will be transferred to one of the manual testers where a more detailed analysis of the defect can be made.

2) PACE and JCE Welded Module Tester. This manual test set will be used to check those modules which cannot be tested by the automatic tester and to trouble-shoot those modules found to be defective by the automatic tester.

3) PACE and JCE Unit Tester. This manual test set is capable of testing both the function boards and the completed units of the PACE and JCE subsystems.

4) Central Timer Tester. This manual test set is used for checking both the welded modules and the complete unit of the central timer subsystem.

5) Telemetry and Command Module Tester. This manual test set is used for the checkout of those welded modules associated with the telemetry encoder and command decoder subsystems.

6) Telemetry and Command Unit Tester. This manual test set is used for function board checkout and unit testing of the telemetry encoder and command decoder subsystem.

7) Logic Board Tester. This manual test set is used to checkout the cards which comprise the digital portion of the ground control equipment.

8) Ground Control System Tester. This manual test set is used to checkout the telemetry processor, command generator, and synchronous controller at the subsystem level.

Module delivery requirements may make it necessary to duplicate one or more of the manual testers.

The following is a preliminary list of commercial equipment which will be used in the test sets. The design of some of the testers is not advanced enough to make a listing of the equipment feasible. All of the commercial equipment will be rack mounted in consoles and will have front panels painted the standard light green Syncom color so that their exclusive use for Syncom project is assured.

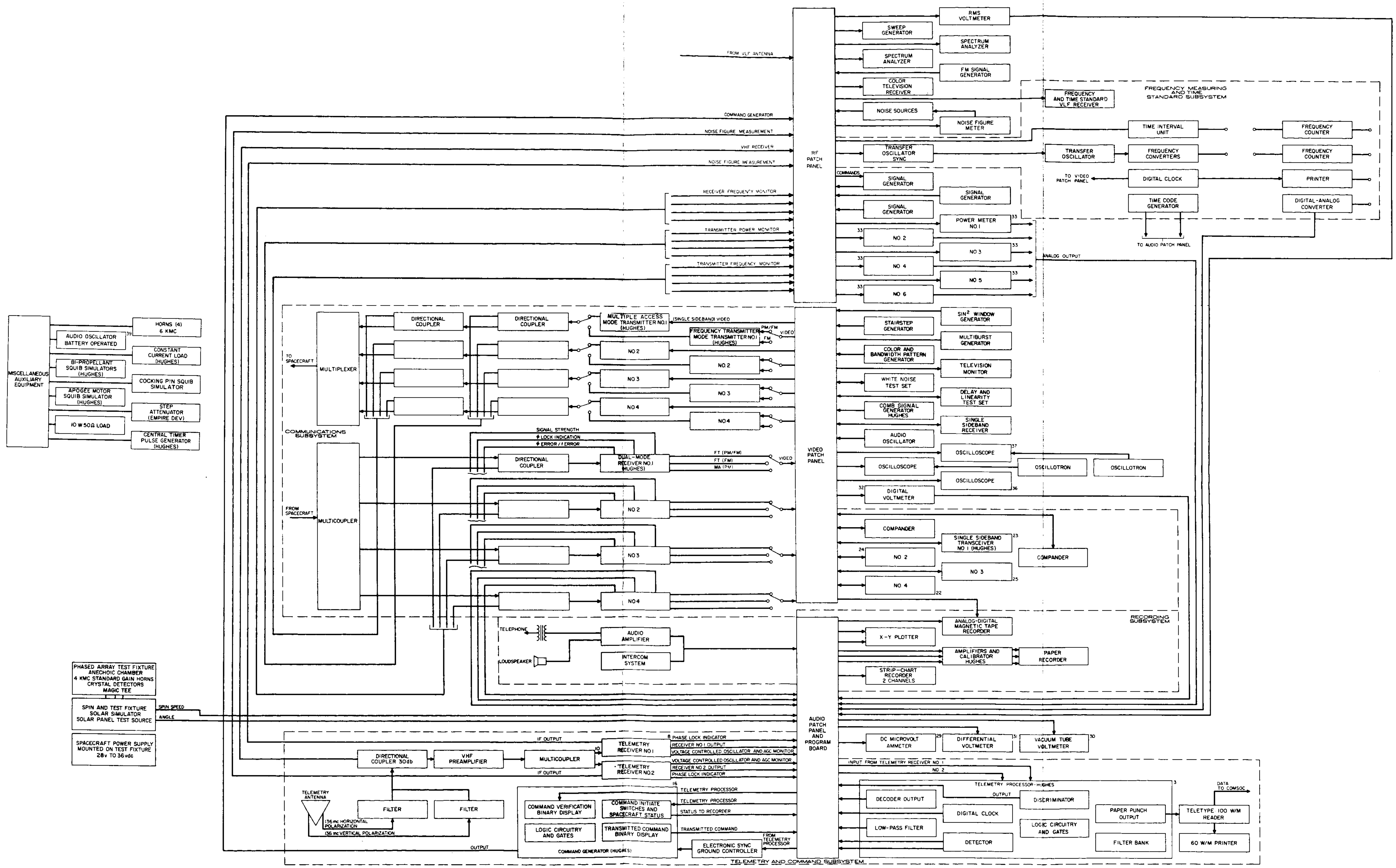


Figure 7-1. Spacecraft Test Station Diagram

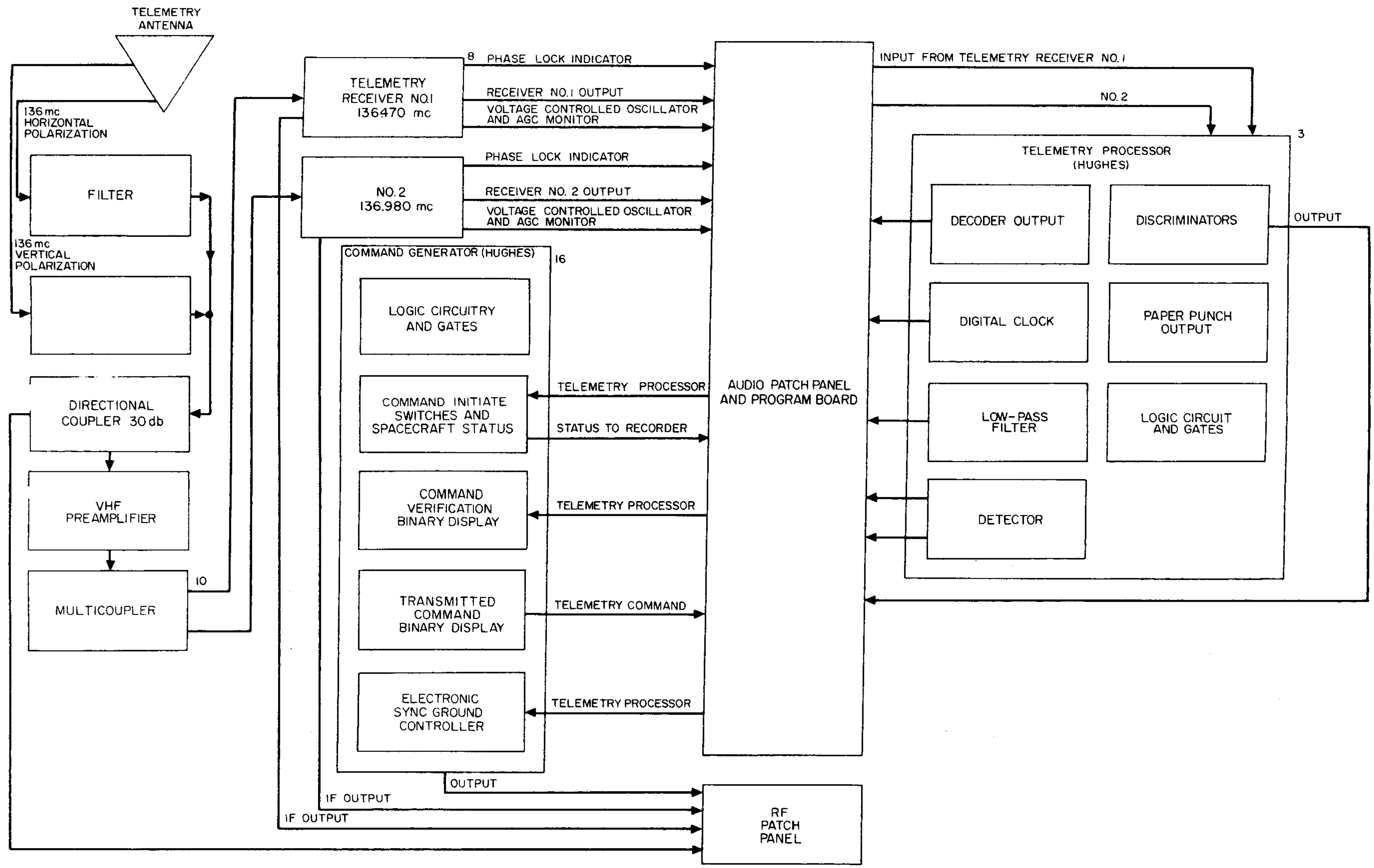


Figure 7-2. Telemetry and Command Subsystems

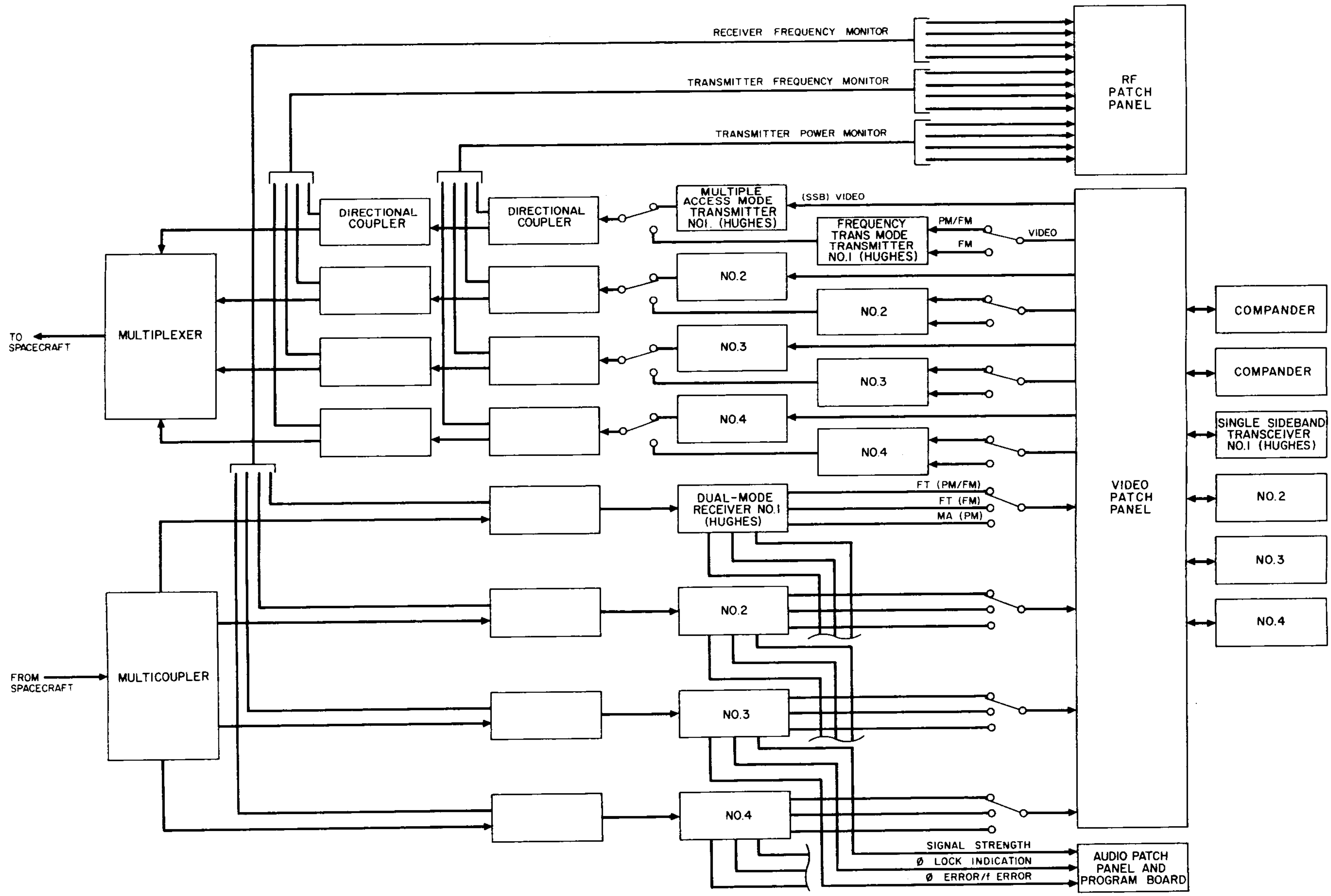


Figure 7-3. Communications Subsystem

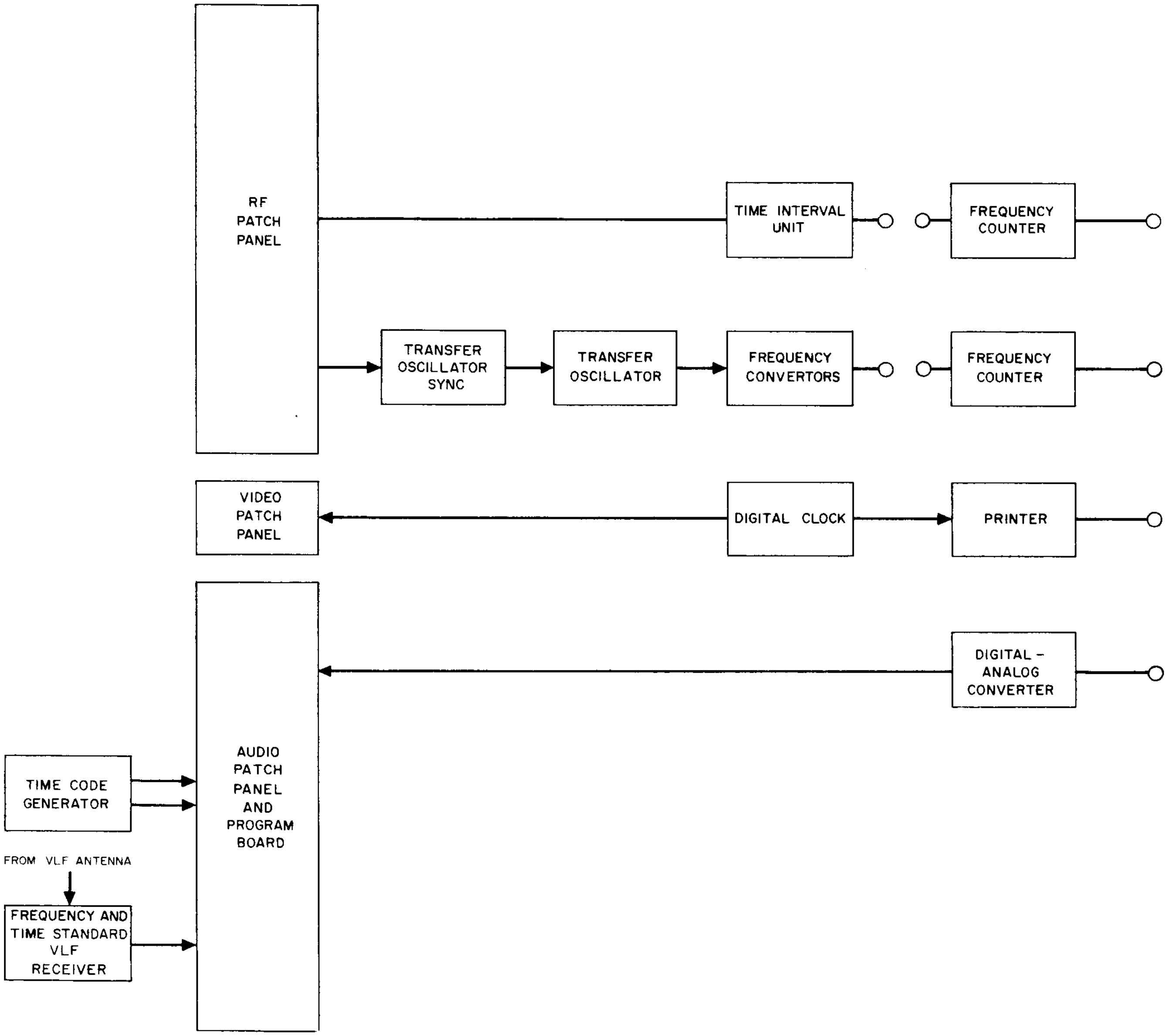


Figure 7-4. Frequency Measuring and Time Standard Subsystem

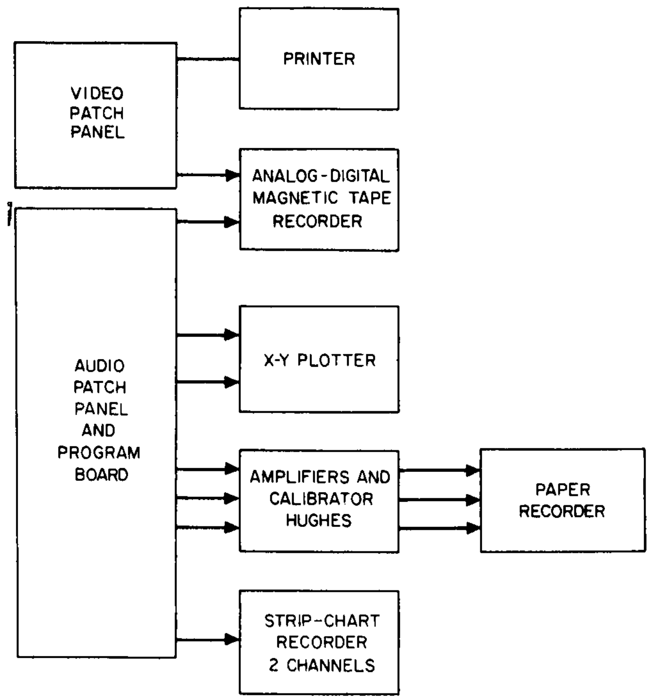


Figure 7-5. Recording Subsystem

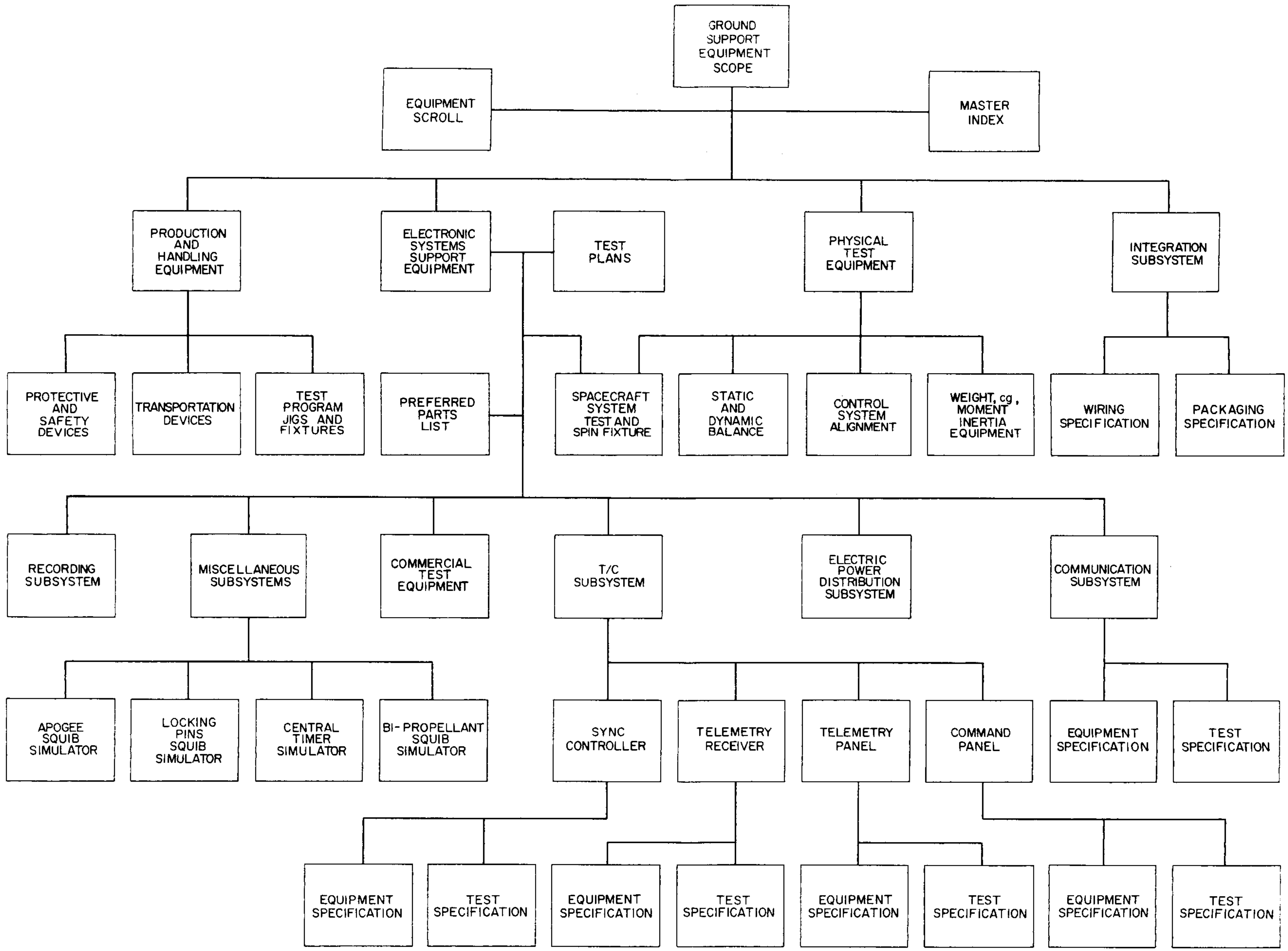
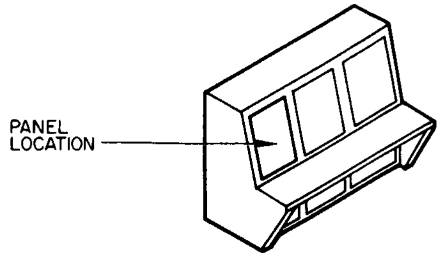
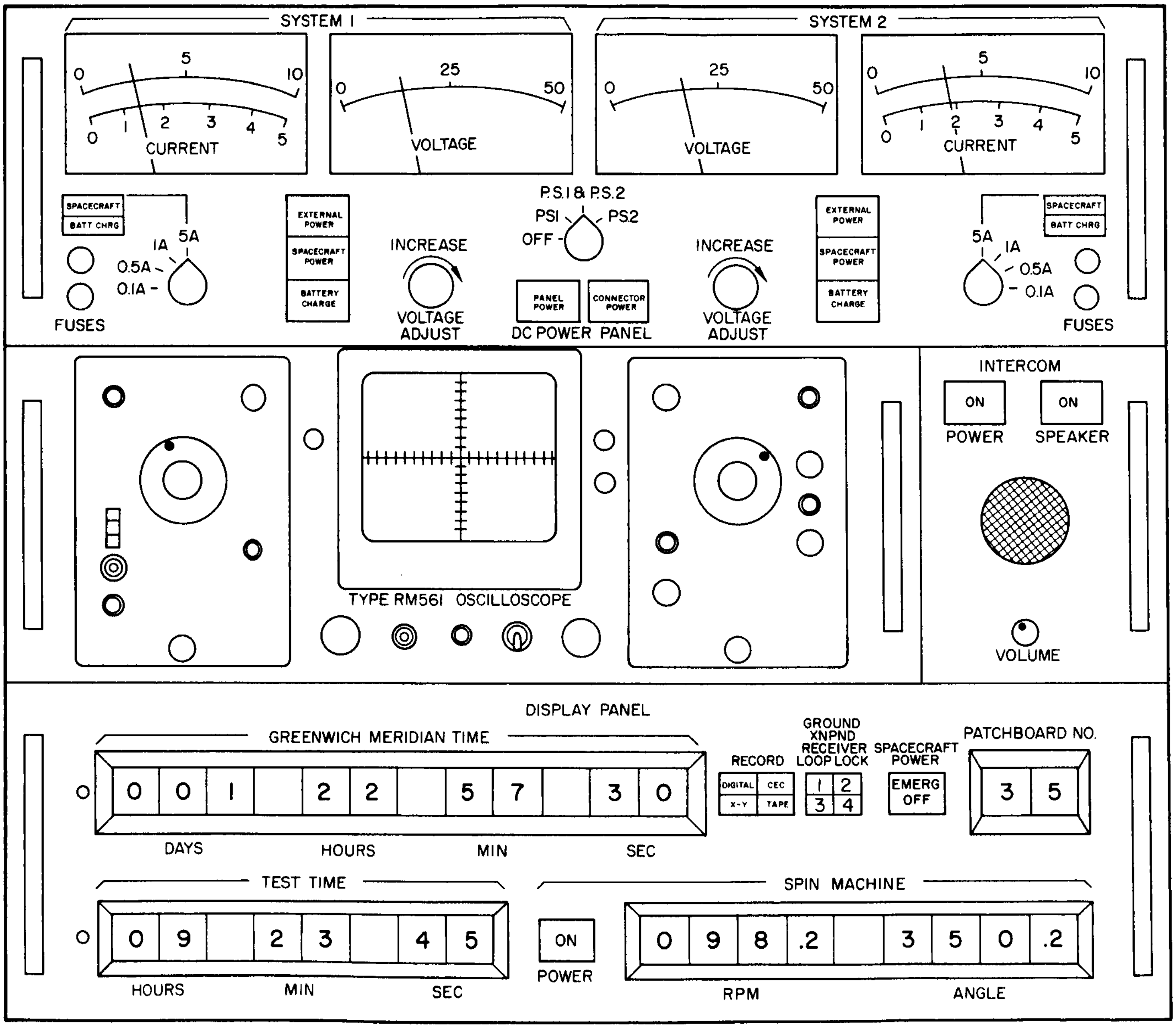


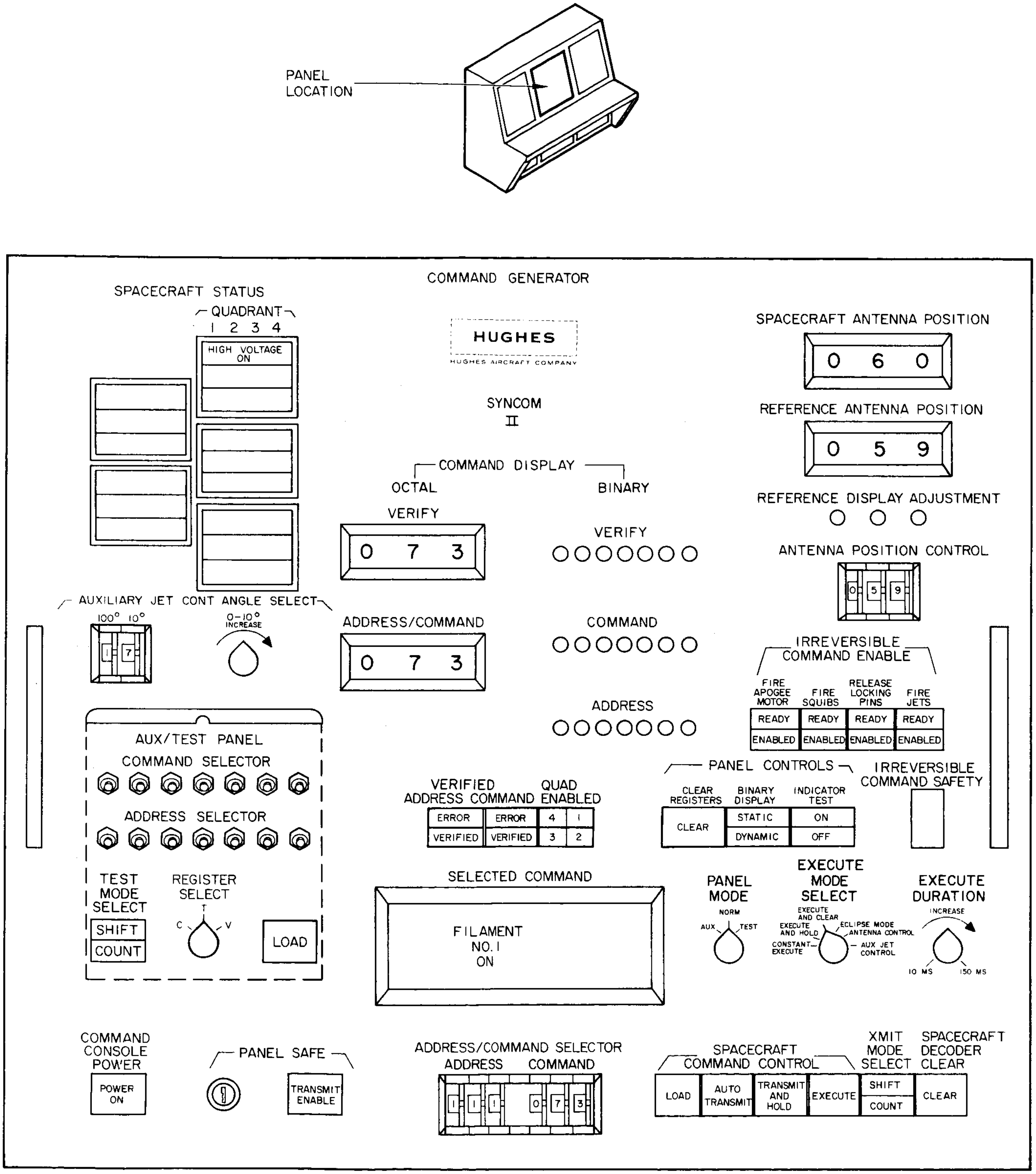
Figure 7-6. Ground Support Equipment Specification Tree





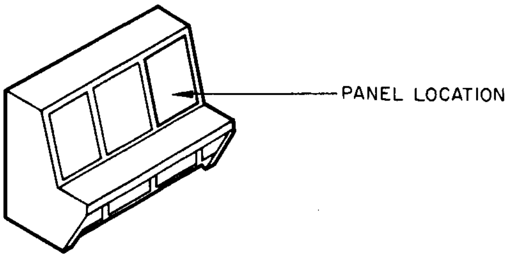
a) DC power panel

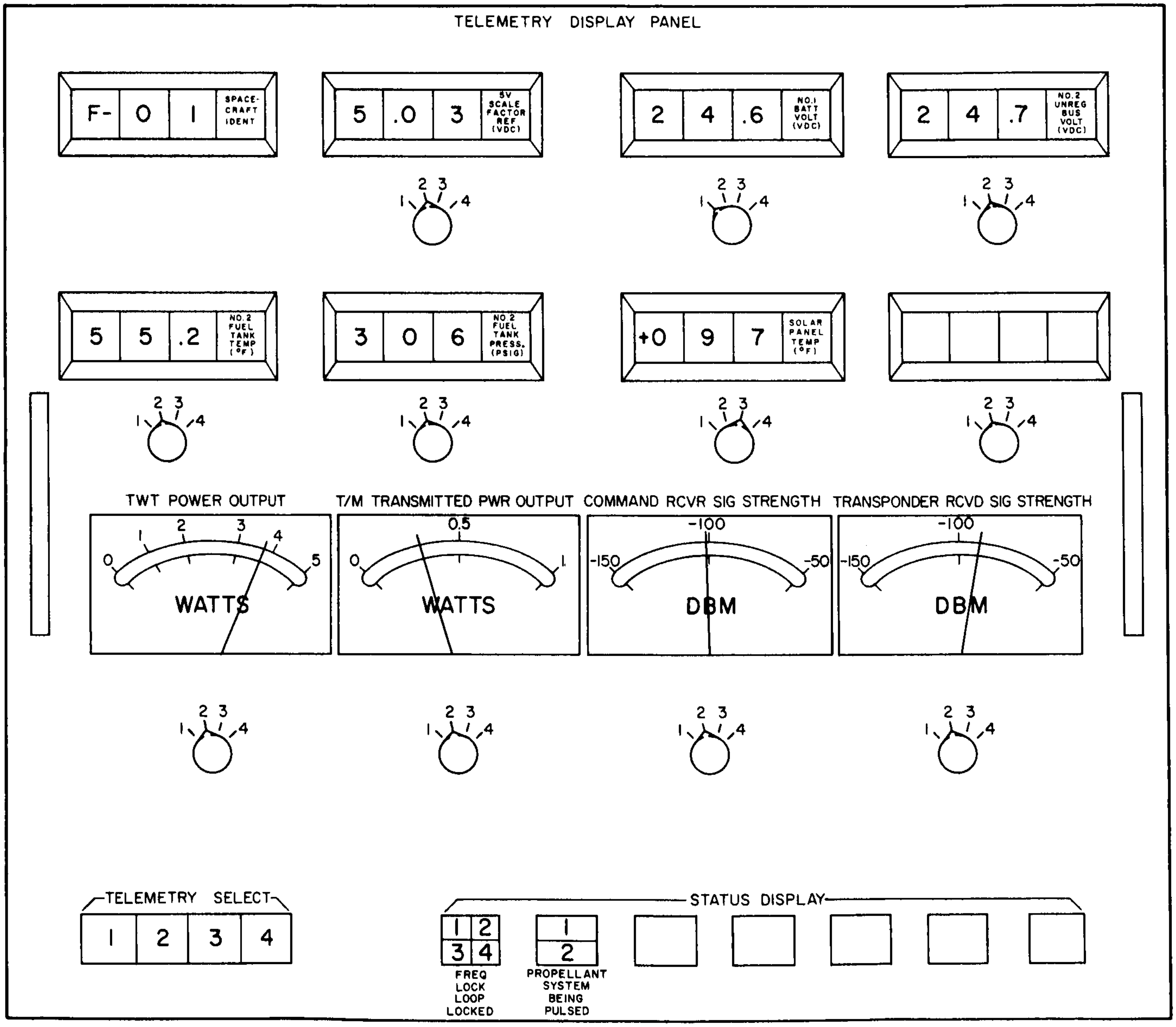
Figure 7-7. Command Console Panels



b) Command generator

Figure 7-7 (continued). Command Console Panels





c) Telemetry display panel

Figure 7-7 (continued). Command Console Panels

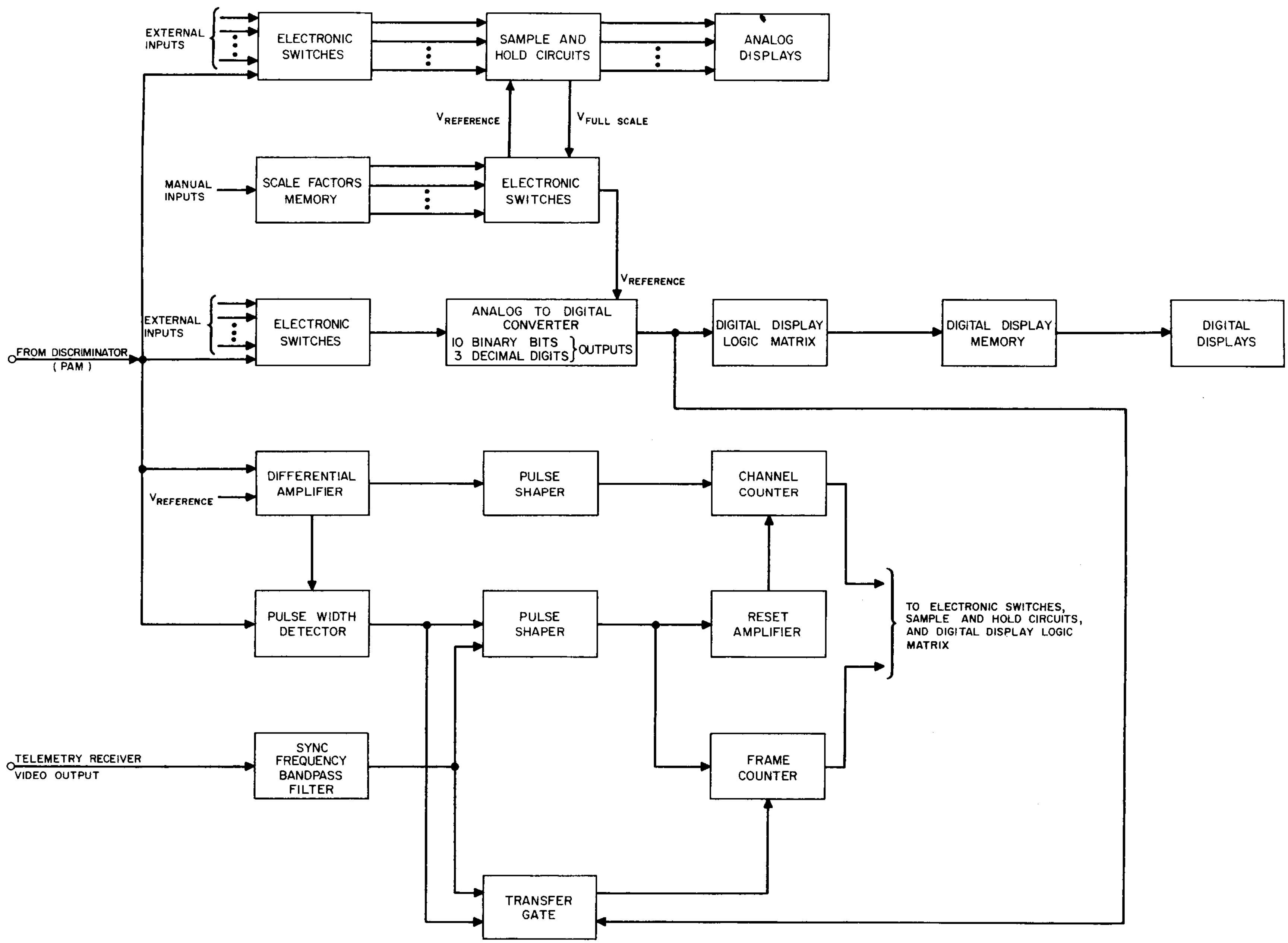


Figure 7-8. Telemetry Processor

PACE and JCE Welded Module Tester

1. Oscilloscope, Tektronix, Type RM35A
   Plug-in, Tektronix, Type M
   Plug-In, Tektronix, Type D
2. Voltmeter, Fluke, Model #825A/AB
3. Voltmeter, H-P, Model #Y94-410BR
4. Pulse Generator, Rutherford, Model #B-7F
5. Function Generator, H-P, Model #202-A
6. Oven, Delta Design, Model #1060A
7. Power Supply, Harrison Lab, Model #6343A (4)
8. Power Supply, Harrison Lab, Model #6346A (6)
9. Power Supply, Harrison Lab, Model #6347A (2)
10. Console, Electronic Enclosures, Model #02-413

PACE and JCE Unit Tester

1. Oscilloscope, Tektronix, RM35A
   Plug-in, Tektronix, Type M
   Plug-in, Tektronix, Type D
2. Amplifier and Current Probe, Tektronix, Model #015-030
3. Oven, Delta Design, Model #1060A
4. Power Supplies, Transpac, TR18P5R (12)
5. Console, Electronic Enclosures, Model #02-413

Central Timer Tester

1. Oscilloscope, Tektronix, RM35A
   Plug-in, Tektronix, Type M
2. Counter, H-P, Model #5253L
   Plug-in, H-P, Model #5262A
3. Amplifier and Current Probe, Tektronix, Model #015-030
4. Voltmeter, H-P, Model #Y94-410BR
5. Console, Electronic Enclosures, Model #02-413
6. Power Supply, Harrison Lab, Model #6347A (3)

Telemetry and Command Module Tester

1. Oscilloscope, Tektronix, Type RM35A
   Plug-in, Tektronix, Type CA
   Plug-in, Tektronix, Type D
2. Oven, Delta Design, Model #1060A
3. Voltmeter, Fluke, Model #825A/AB
4. Voltmeter, H-P, Model #Y94-410BR
5. Console, Electronic Enclosures, Model #02-413

Telemetry and Command Unit Tester

1. Oscilloscope, Tektronix, RM35A
   Plug-in, Tektronix, Type M
2. Console, Electronic Enclosures, Model #02-413
3. Pattern Generator, Datapulse, Model #200M
   Plug-in, P901
4. Voltmeter, H-P, Model #Y94-410BR

Communications Test Equipment

The communications RF test equipment (Figure 7-9) consists of the following major components.

1) Four Frequency Translation Transmitters

2) Four Frequency Translation Receivers

3) Four Multiple Access Transmitters

4) Four Multiple Access Receivers

The block diagram shows only one of the transmitters and receivers listed above. This corresponds to the test equipment required to check out one spacecraft transponder. Except for frequency differences as indicated in the block diagram, the test equipment for all four transponders is identical. The following paragraphs describe equipment in more detail.

Frequency Translation Transmitter

The block diagram shows two modulation methods that are being considered for the frequency translation transmitters.

The first method accepts wide band video at J1 which frequency modulates the output of a voltage-controlled oscillator (VCO). Negative feedback from a discriminator centered at the nominal frequency of the VCO provides frequency stability.

The second modulation method accepts wide band video at J2. The video is passed through an equalizing network which tends to flatten the amplitude of the video spectrum. The modified video is then applied to a phase modulator which produces an output signal which may be considered to be a composite of phase and frequency modulation due to the spectrum shaping property of the equalizing network. The equalizing network tends to compensate for the disproportionately large carrier phase deviation caused by the relatively strong low frequency components of TV video as compared with that caused by the weaker high frequency components. The overall objective of this method is to provide a good quality TV picture at signal-to-noise levels that would cause objectionable distortion of the high frequencies when using conventional modulation techniques.

The modulated signal, whether it be FM or PM/FM, is multiplied by 32. This transfers the information into the 6 gc range and increases the modulation index. The power level is increased by a TWT and suitable bandpass filters remove the unwanted frequencies. The final RF signal (J3) has a 25 mc bandwidth and a maximum level -3dbm.

Frequency Translation Receivers

The input (J4) to each frequency translation receiver is 4 gc FM (or PM/FM) signal from the spacecraft. The received signal is filtered in a bandpass filter and amplified by two isolated TWT's. A divide-by-two network puts the frequency of the received signal in the 2 gc range. Another TWT and an attenuator provide the required signal level at the input of a divide-by-32 network. The output of the divide-by-32 network is an IF signal of approximately 60 mc which is fed into a power splitter. Both the divide-by-two and the divide-by-32 circuits are derived from spacecraft hardware. Each is a modified version of the spacecraft multiplier chains driver backwards.

Again two demodulation schemes are shown on the block diagram to account for the two modulation schemes described for the transmitter. If the received signal is frequency modulated only, the IF signal is passed through a limiter to remove amplitude variations. The video is recovered at the output of a wideband discriminator (J5).

If the received signal is a combination of PM and FM, the IF signal is passed through a limiter, phase detector, and spectrum restoring network. This network restores the modified video spectrum to its original shape at J6. (The video spectrum was purposely equalized prior to phase modulation.) It is possible to demodulate with a phase detector because the high modulation index of the received signal has been divided by 64.

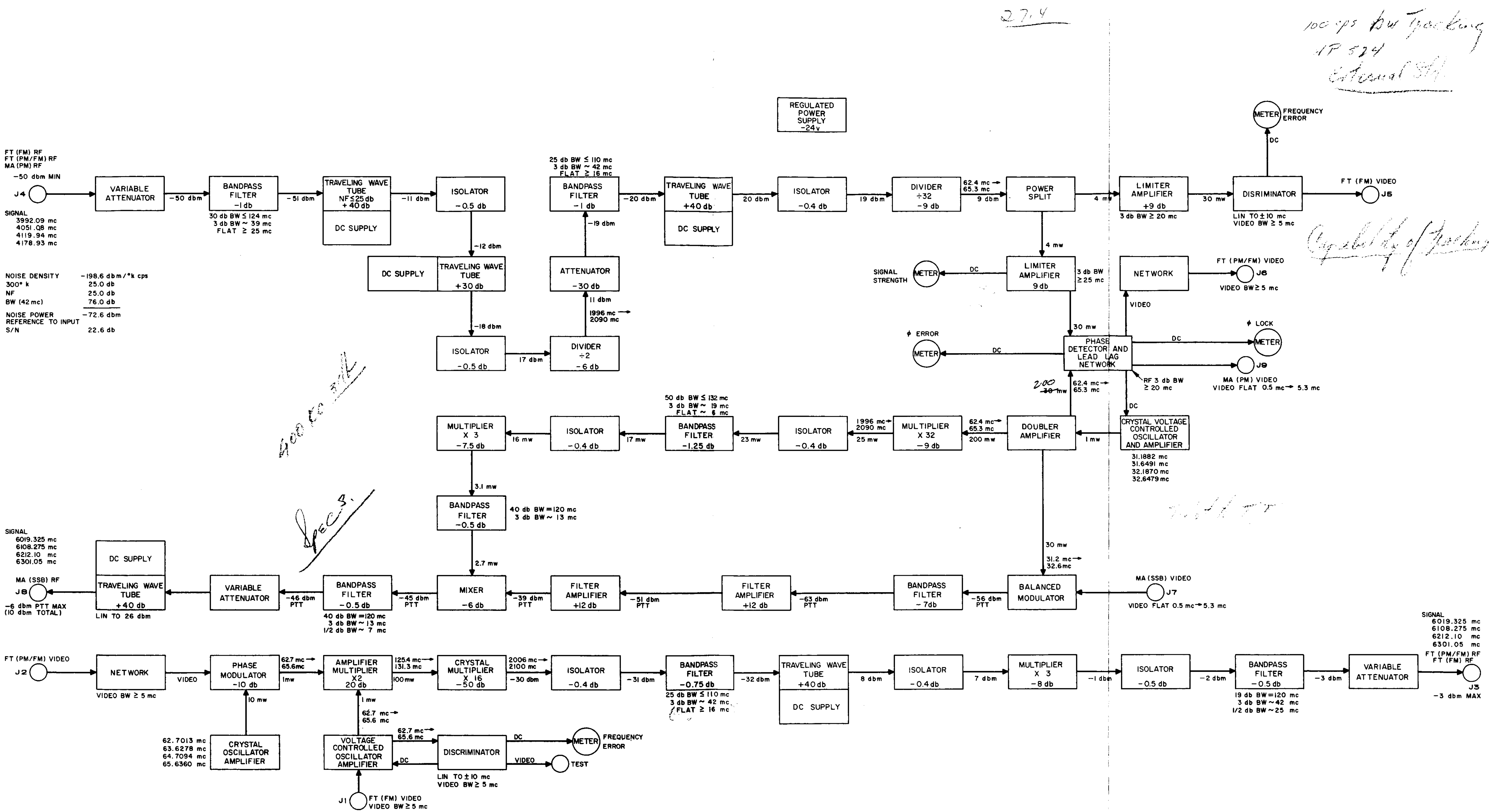


Figure 7-9. Preliminary Block Diagram, Syncom II Transponder RF Test Set

Multiple Access Transmitters

Each multiple access transmitter accepts video at J7. (The video is a composite base band of frequency multiplexed voice channels having a flat bandwidth of 4.8 mc.) The video is applied to a balanced modulator which produces a double sideband spectrum centered about a suppressed carrier of approximately 30 mc. The lower sideband is removed and the signal level increased by subsequent filters and amplifiers. A 6 gc mixer translates the upper sideband to the required transmitter frequency. The difference frequencies and the 6 gc mixing signal are rejected by a bandpass filter while the sum frequencies are amplified by a TWT operating at low power in the linear region.

The 30 mc and 6 gc carriers are derived from a VCO which is phase-locked to the spacecraft return signal.

The final RF signal (J8) is single sideband modulated. It has a bandwidth of 4.8 mc and a maximum power level of -6 dbm per test tone.

Multiple Access Receivers

The input (J4) to each multiple access receiver is a 4 gc phase modulated signal from the spacecraft. A 60 mc IF signal is derived in exactly the same manner as that of the frequency translation receiver. A limiter then removes amplitude variations from the IF signal prior to phase detection.

Two outputs are obtained from the phase detector. The first is the multiple access video which is recovered at J9. The second output is a dc voltage which controls a VCO. Since the output of the VCO provides a reference signal for the phase detector, the VCO becomes phase-locked to the received signal. The 30 mc and 6 gc carriers required for the multiple access transmitter are derived from the VCO.

Four phase detector networks have been built and tested during this report period. From these tests, the design was determined which provides the most nearly linear and balanced output. This design includes a variable resistor which provides a dc balance. (See Figure 7-10.)

Data Flow Chart

Figure 7-11 shows the proposed data flow of the data recording facilities. Most major test and all recording equipments are shown in separate blocks. Additional test data signals are received from miscellaneous equipments and the spacecraft.

Data is recorded on strip chart recorders, X-Y plotters, and printers for quick look purposes and on magnetic tape recorders for a permanent record. Patching for individual tests is accomplished by exchanging pre-patched boards, one analog and one digital for each test.

Digital voltmeter and millimeter with binary coded decimal (BCD) outputs were chosen for compatibility with outputs of frequency counters and with digital recording equipment. The analog data signals are converted to BCD in the analog-to-digital converters. These parallel BCD data signals are accepted by the printer. For recording on magnetic tape, the parallel BCD signals are converted to serial data in the parallel-to-serial converters and recorded serially, one data channel per track. Two digital tracks with NRZ modules are available and one direct record track is used for clock pulses. Spacecraft status signals are recorded using a FM module. Digital data recorded on tape can be displayed on appropriate display equipment after data has been converted into BCD format by the serial-to-parallel converters.

Command signals, undemodulated telemetry signals, and time code are recorded on the direct record tracks with voice recorded on the auxiliary track (track A). Digital-to-analog converters are available to convert digital data into analog data for recording on the analog equipment.

Spacecraft position and spacecraft spin speed signals are pulses generated at the spin machine. Pulses are generated as a function of spacecraft position at a rate proportional to the spacecraft's spin speed.

Preliminary Equipment Arrangement

The layout shown (Figures 7-12 and 7-13) is the first attempt to define the test position to be used for the testing of Syncom II spacecraft. The test position is composed of 16 bays of equipment and a 3-bay control and monitoring console. The 16 bays of equipment are divided into three main groups. These groups are RF, video, and audio. The RF group contains equipment operating in the frequency range of 140 mc and up. The video group equipment operates in the video to 140 mc frequency range. The audio group equipment operates in the dc-to-video frequency range. Each of the above groups has a patch panel. The audio group also has a patch panel programmer.

The RF and audio group have 6 bays of equipment each, and the video group has 4 bays of equipment. The equipment used for testing purposes is placed as near the appropriate patch panel as possible. The RF and video patch panels are located near each other because of common connections. The audio group is separated from the RF and video group by the control console.

Liberal allowance is provided for the patch panels and storage. Shelves and intercom alternate throughout the test position to allow sufficient writing surface and communication jacks.

Test Equipment Requirements

The ground support equipment requirements have been detailed on a master index similar to that used for spacecraft equipment control. The following master index lists the present makeup of the ground support equipment master index.

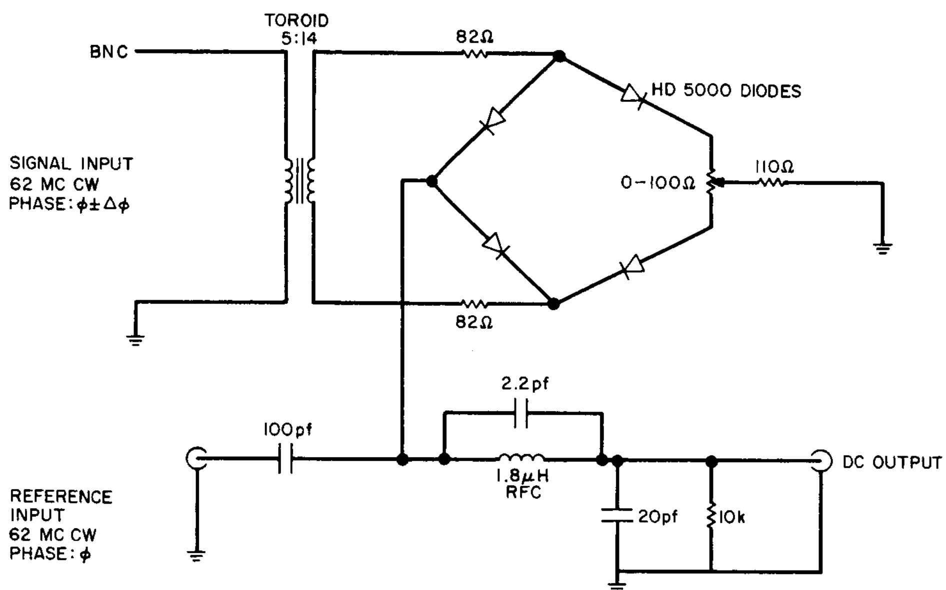
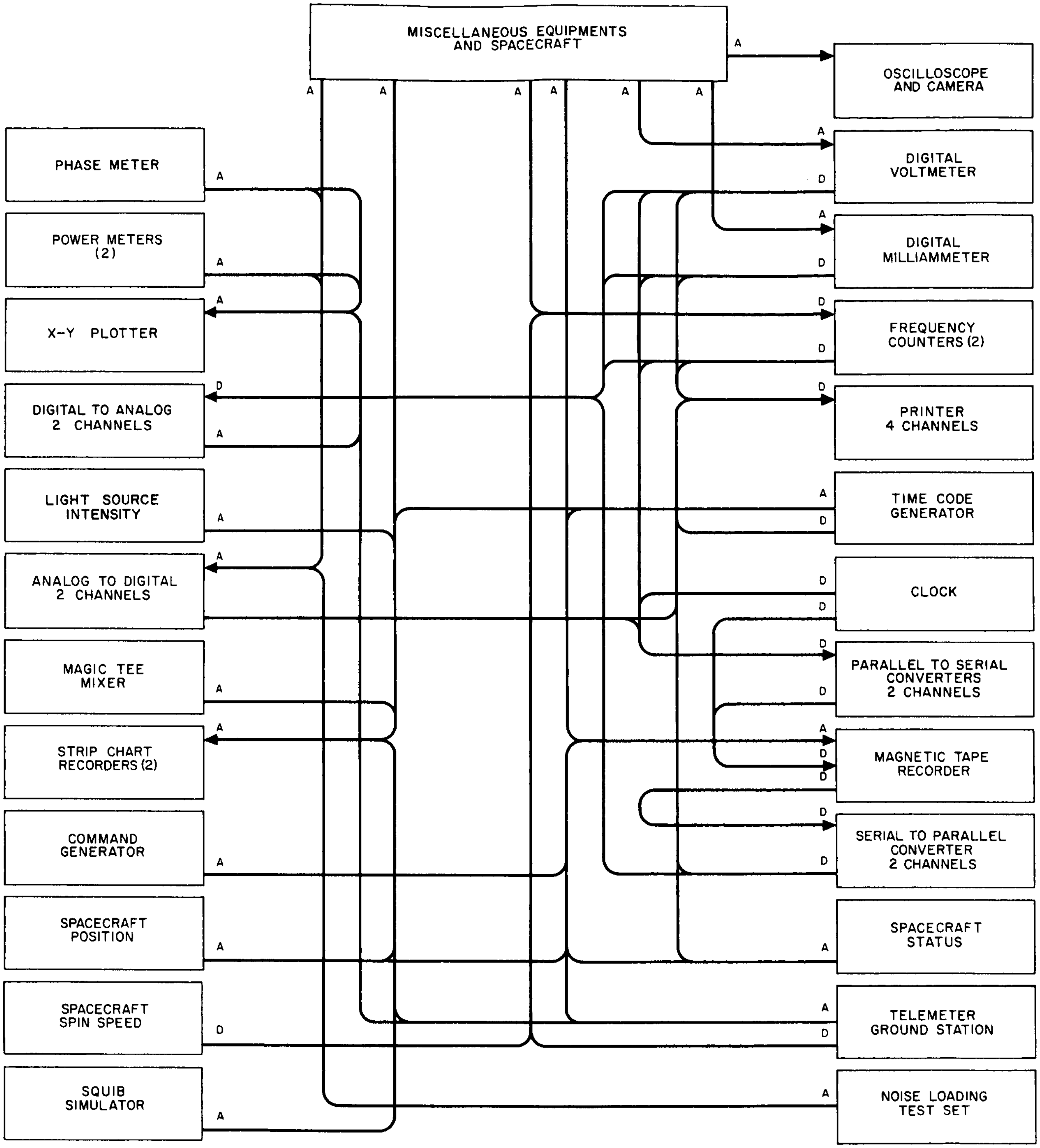


Figure 7-10. Phase Detector



A CONNECTED VIA ANALOG PATCH PANEL
D CONNECTED VIA DIGITAL PATCH PANEL
PATCHING PERFORMED BY EXCHANGING PRE-PATCHED BOARDS

Figure 7-11. Data Flow Chart and Data Recording Facilities

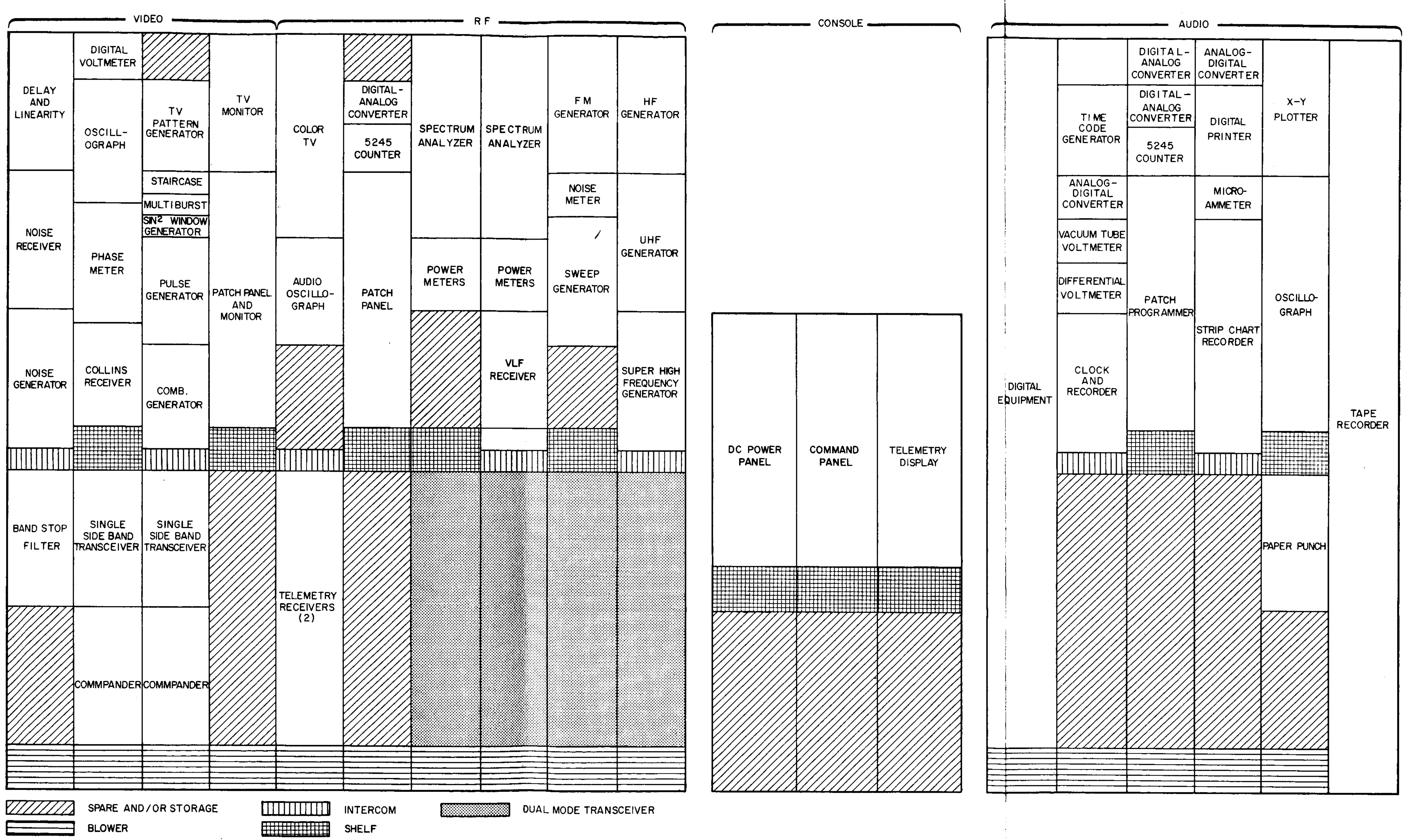


Figure 7-12. General Equipment Arrangement

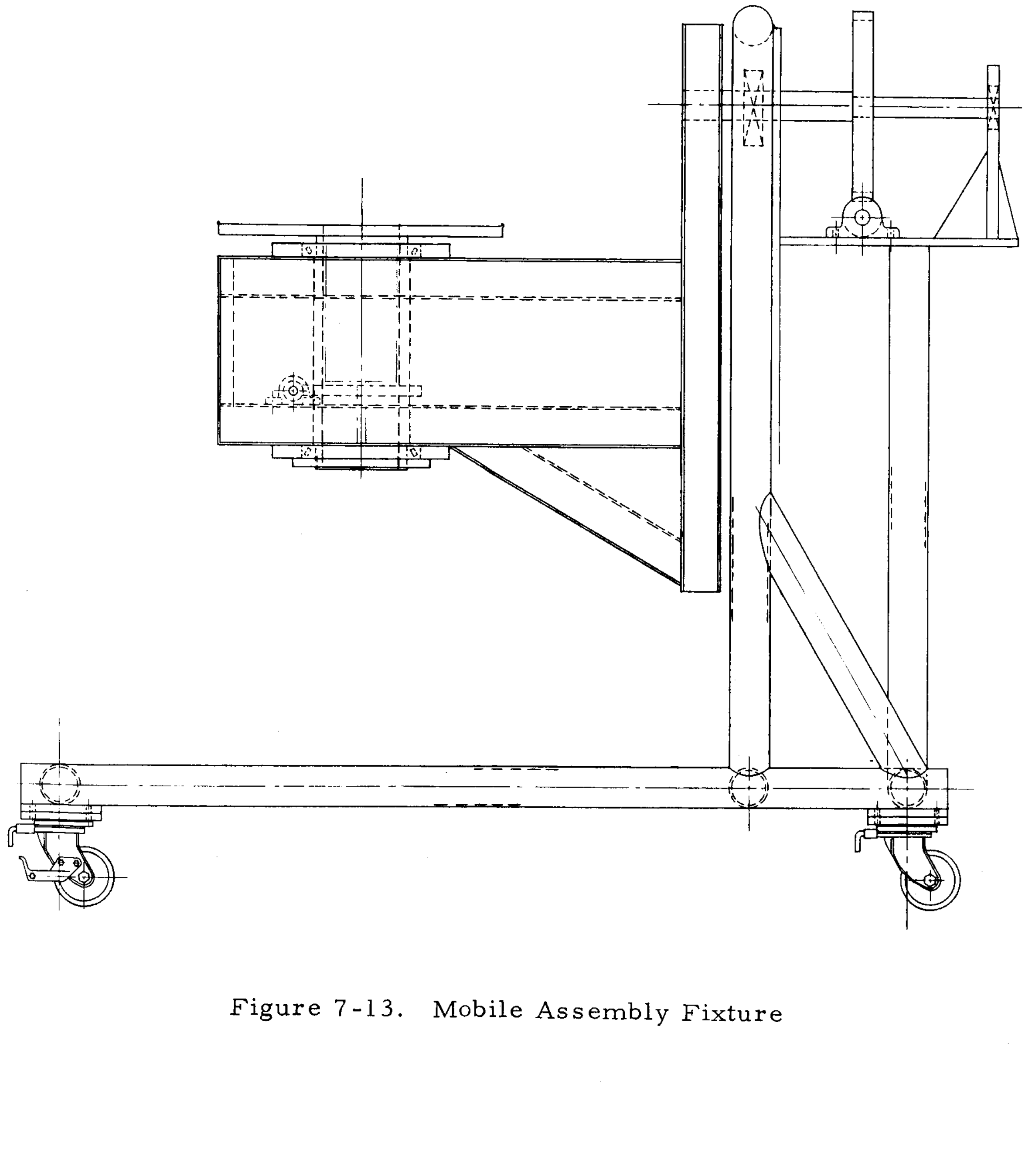


Figure 7-13. Mobile Assembly Fixture

MASTER INDEX (I)	ELECTRONIC GROUND SUPPORT EQUIPMENT, SYNCOM II 475500-100 PRODUCT NAME DESIGNATION	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 82577	2 SH NO.	C REV	MI 475500-100 MASTER INDEX NO.

REV LTR	CONTROL ITEM PART NUMBER		QTY*	CONTROL ITEM NAME	COMMERCIAL DESIGNATION	NOTES
	475510	100	1	COMMUNICATION SUBSYSTEM		Major Control Item
	475540	100	1	Dual-Mode Receiver #1		
		101	1	Dual-Mode Receiver #2		
		102	1	Dual-Mode Receiver #3		
		103	1	Dual-Mode Receiver #4		
			1	Multiplexer for Dual Mode Receivers	To be selected	
	475541	100	1	Dual-Mode Transmitter #1		
		101	1	Dual-Mode Transmitter #2		
		102	1	Dual-Mode Transmitter #3		
		103	1	Dual-Mode Transmitter #4		
			1	Multiplexer for Dual-Mode Transmitters	To be selected	
	475542	100	1	SSB Transceiver #1 (Baseband)		
		101	1	SSB Transceiver #2 (Baseband)		
			1	TV Monitor	Kintel GRM-17-R	
			1	Color TV Receiver	To be selected	
			1	SSB Receiver	Collins 51C	
	475543	100	2	Compander		
	475511	100	1	TELEMETRY & COMMAND SUBSYSTEM		Major Control Item
	475560	100	1	Command Signal Generator		
	475561	100	1	Electronic Synchronous Ground Controller		

*Per set of equipment

MASTER INDEX (I)	ELECTRONIC GROUND SUPPORT EQUIPMENT, SYNCOM II 475500-100 PRODUCT NAME/DESIGNATION	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 82577	SH NO. 3	REV C	MI 475500-100 MASTER INDEX NO.

REV LTR	CONTROL ITEM PART NUMBER		QTY	CONTROL ITEM NAME	COMMERCIAL DESIGNATION	NOTES
	475562	100	1	Telemetry Processor		
			1	Telemetry Receiver #1	Space General 136.470 mc	
			1	Telemetry Receiver #2	Space General 136.980 mc	
	475563	100	1	Telemetry Receiver Diplexer		
	475512	100	1	INSTRUMENTATION SUBSYSTEM		Major Control Item
			1	Audio Patch Panel	Trompeter Electronics	
			1	RF Patch Panel	Trompeter Electronics	
			1	RMS Voltmeter	Ballatine 320	
			1	DC Microvoltmeter	Kintel 203AR	
			1	VTVM	HP 410B	
			1	Differential Voltmeter	To be selected	
			1	Digital Voltmeter	HP 405CR	
			4	Power Meter	HP 431B	
			1	Phase Meter	AD-YU Electronics	
			1	Oscilloscope	Tek 555A	
			1	Oscilloscope	RM 45A	
			1	Oscilloscope	RM 15	
			2	Pre-Amp	Tek Type C-A, Dual Trace*	*These Pre-Amps to
			1	Pre-Amp	Tek Type H, Wideband, High Gain*	be used with RM 45A
			1	Pre-Amp	Tek Type L, Fast Rise, High Gain*	and Tek 555A

MASTER INDEX (I)	ELECTRONIC GROUND SUPPORT EQUIPMENT, SYNCOM II 475500-100 PRODUCT NAME DESIGNATION	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 82577	SH NO. 4	REV C	MI 47550C-100 MASTER INDEX NO.

REV LTR	CONTROL ITEM PART NUMBER	QTY	CONTROL ITEM NAME	COMMERCIAL DESIGNATION	NOTES
		1	Pre-Amp	Tek Type L Fast Rise High Gain	These Pre-Amps to be used with RM 45A and Tek 555A
		1	Pre-Amp	Tek Type M, Four Trace	
		2	Scope Camera	Beatle Coleman Mark II D	
		1	Mobile Scope Cart	Tek 50C/53A	To be used with Tek 555A
		1	Audio Oscillator	HP 200 TR	
		1	Audio Oscillator	HP 204B	
		1	HF Signal Generator	HP 606AR	
		1	VHF Signal Generator	HP 608DR	
		1	SHF Signal Generator	HP 618BR	
		1	FM Signal Generator	Marconi TF 1066B/1	
		1	Stairstep Generator	Telech. 3502A1	
		1	Sin^2 Window Generator	Telech. 3503A1	
		1	Sweep Generator	To be selected	To be used with HP 618BR
		1	White Noise Test Set	Marconi OP1249B/R	
		1	Noise Figure Meter	HP 340BR	
		1	Noise Source	To be selected	
		1	Color, Black and White TV Pattern Generator	To be selected	
		1	Frequency and Time Standard ULF Receiver	Gertsh	
		2	Spectrum Analyzer	Lavoie LA 18M	
		4	TWT Amplifier	To be selected	
		1	Delay & Linearity Test Set	To be selected	

MASTER INDEX (I)	ELECTRONIC GROUND SUPPORT EQUIPMENT, SYNCOM II 475500-100 PRODUCT NAME DESIGNATION	HUGHES AIRCRAFT CO. CULVER CITY, CALIF. CODE IDENT 82577	5 SH NO.	C REV	MI 475500-100 MASTER INDEX NO.

REV LTR	CONTROL ITEM PART NUMBER		QTY	CONTROL ITEM NAME	COMMERCIAL DESIGNATION	NOTES
			1	Step Attenuator	Empire Devices	
			1	Frequency Converter	HP 5251	
			1	Microwave Frequency Counter	HP 2590A	
			1	Frequency Converter	HP 5253A	
			2	Frequency Counter	HP 5245L	
			1	Multiburst Generator	Telech.	
			1	Time Interval Unit	HP 5262A	
	475513	100	1	RECORDING & DATA SUBSYSTEM		
			1	Magnetic Tape Recorder	To be selected	
			1	Recording Oscillograph	CEC 5-119	
			1	Strip Chart Recorder (2 channels)	To be selected	
			1	Recorder Calibration Unit	To be selected	
			1	X-Y Plotter	To be selected	
			1	TTY 100 W/M Reader	To be selected	
			1	TTY 60 W/M Printer	To be selected	
			1	Paper Tape Punch	Cy-2542	
			1	Digital Recorder	HP 560AR	
			1	Digital Printer	HP 563A	
			1	Digital Clock	HP 570AR	
			1	Time Code Generator	E.E. Corp.	
			2	Analog-Digital Converter	To be selected	
			3	Digital Analog Converter	HP 580A	

MASTER INDEX (I)

ELECTRONIC GROUND SUPPORT EQUIPMENT, SYNCOM II
475500-100
PRODUCT NAME/DESIGNATION

HUGHES AIRCRAFT CO.
CULVER CITY, CALIF.
CODE IDENT 82577

6 SH NO. | C REV | MI 475500-100 MASTER INDEX NO.

REV LTR	CONTROL ITEM PART NUMBER		QTY	CONTROL ITEM NAME	COMMERCIAL DESIGNATION	NOTES
	475580	100	1	Electronic Psi Simulator		
	475581	100	1	Bipropellant Squib Simul.		
	475582	100	1	Pulse Generator, Apogee Motor Timer		
	475583	100	1	Remote Switching Panel		

8. SPACECRAFT HANDLING EQUIPMENT

HOLDING FIXTURE, AFT SEGMENT

This fixture, designed originally for the T-1 model, was redesigned. The 12-inch head was replaced with a 20-inch diameter index head. The drawing is now being checked.

MOBILE ASSEMBLY FIXTURE

The cantilever structure to mount the spacecraft and its attach plate for axial rotation has been designed. Design work is continuing on the framework to support and rotate the cantilever structure (see Figure 8-1).

SYSTEM TEST AND SPIN FIXTURE

Design effort has been concerned with general arrangement and determination of power requirements based on windage estimates of the spacecraft, bearing friction of the fixture, and the spin stability requirements. Discussions have been made with vendors of electrical motors to discuss motor speed-control systems and noise levels. These discussions will continue, and the products evaluated before a definite selection of power components will be incorporated in the fixture design. Similar contacts with vendors have been made and are continuing on slip rings, coaxial rotary joints, and the fixture chassis components (wheels and jacks). Design studies are in progress on quick-disconnect methods of attaching the fixture mounting disk to the apogee motor mounting surface on the structure. Methods are under design study for simple alignment adjustment of this surface to read out to specified tolerances with the Agena mounting surface on the structure.

BLANCHING MACHINE

Design work for the blanching machine is in its early stages with general arrangement studies being conducted. This work is based on the latest mass properties design requirements.

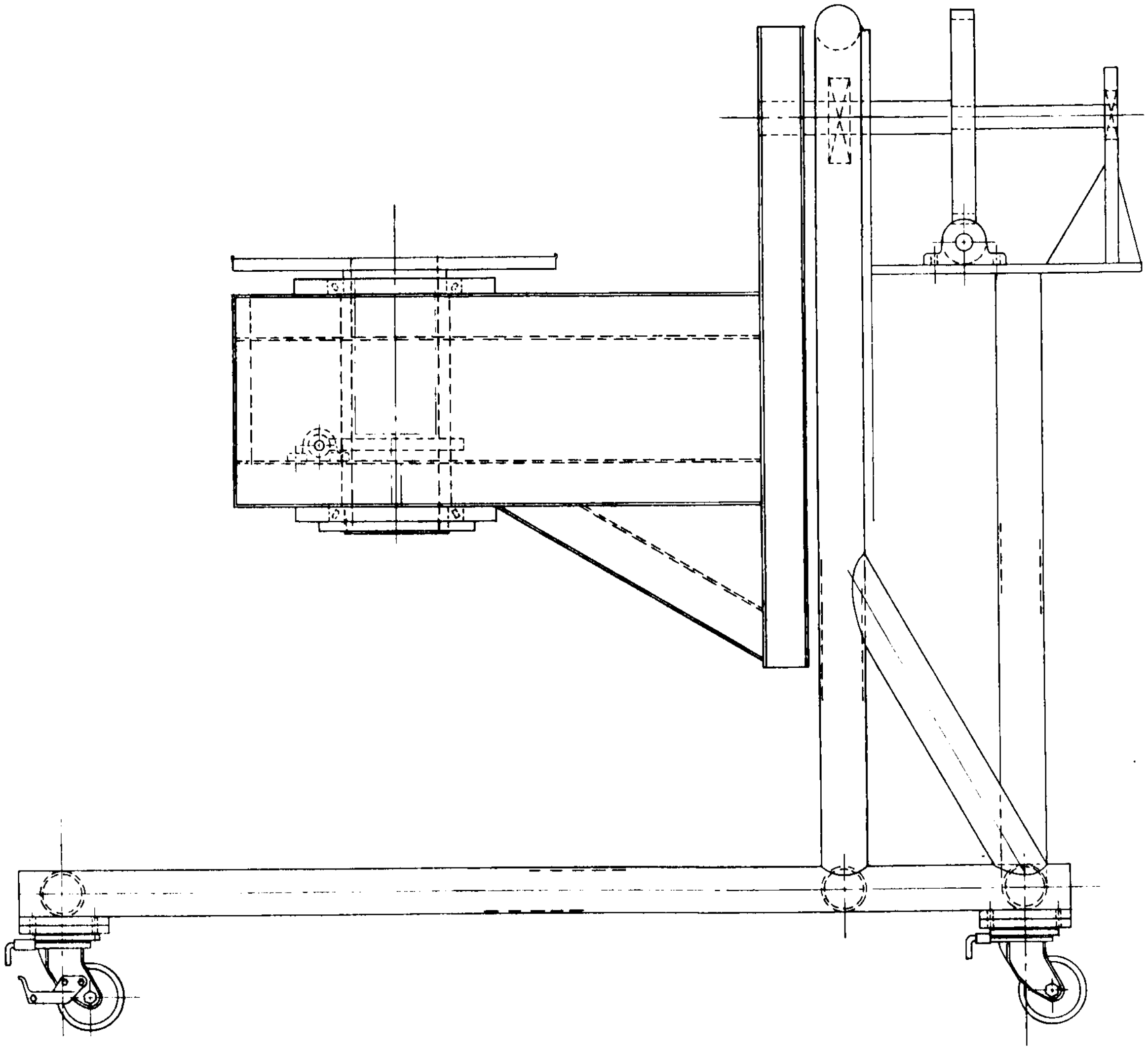


Figure 8-1. Mobile Assembly Fixture

9. NEW TECHNOLOGY

WHIP ANTENNA ORIENTATION

A novel orientation of the telemetry and command whip antennas has resulted from studies for the adaption of the Syncom I antenna design to the Syncom II spacecraft. The particular orientation provides an optimum radiation pattern with an acceptable amount of spin modulation and could be applied to other situations with similar antenna requirements by varying the orientation and number of whips according to the dimensions involved and the operating frequencies.

The Syncom I telemetry and command subsystem consists of two transmitters, two receivers, and a turnstile antenna arrangement of four quarter-wavelength whips spaced at 90-degree intervals around the spacecraft circumference. The transmitters and receivers share the same antennas by frequency diplexing. When this design is applied to the larger Syncom II spacecraft for the operation of a set of two each of the four transmitters and receivers, undesirable 10 db interference minima are produced in the radiation pattern. Therefore, eight whips instead of four are used for each set of two transmitters and receivers. The whips are equally spaced around the circumference of the spacecraft and alternately oriented at approximately 30 degrees and 90 degrees from the positive spin axis. Each set of eight whips is fed through two hybrid baluns with a 45-degree phase difference between adjacent whips. The whips oriented radially are operated independently of those oriented at 30 degrees and the interference minima of the two do not coincide. This arrangement, which is being experimentally verified, reduces the mutual coupling between elements and produces a diversified polarization pattern with the interference minima decreased to approximately ±1 db.

The above technology is not considered to constitute an invention but is an innovation of the state of the art which has potential utility in other applications.

GENERAL

Five other items investigated during the report period are believed to constitute inventions and will be reported separately.

10. PROJECT REFERENCE REPORTS

P. J. Fisher, "Preferred Parts List, Hughes Document 30505-004," 26 June 1963.

H. A. Roth, "Temperatures of Syncom II Axial Reaction Control Nozzles," IDC, 20 June 1963.

"Syncom II Interface Data," LMSC Document A376151, 18 June 1963.

P. W. Michelsen, "Proposed Data Flow Data Recording Facilities," IDC, 24 June 1963.

L. V. Erdle, "Minutes of GSE Meeting for Advanced Syncom Command Generator and Telemetry Processor," IDC, 26 June 1963.

P. R. Broward, "Minutes of Syncom II RCS Interface Meeting," IDC, 21 June 1963.

F. B. Bjorklund, "Syncom II Thermal Control Inputs," IDC, 20 June 1963.

R. A. Houghten, "Syncom II Coordination Control Summary Report," 21 June 1963.

P. S. Dupont, "Revision to Syncom II Battery Specifications," IDC, 20 June 1963.

J. G. Lotta, "Syncom II Spin Balance Machine Design Requirements," IDC, 20 June 1963.

P. E. Norsell, "Increase of Solar Panel Length," 20 June 1963.

P. S. Dupont, "Increase of Solar Panel Length," IDC, 11 June 1963.

M. E. Clay, "Syncom II Solar Panel Development Dynamic Tests," IDC, 17 June 1963.

J. P. Wrzesinski, "Placement of Solar Sensors Relative to the Radial Jets," 20 June 1963.

J.P. Wrzesinski, "Solar Panel Weights," 20 June 1963.

J.M. Ferry, "Coordination Meeting Syncom II RCS Interface," IDC, 14 June 1963.

L.A. Gustafson, "Control Items Located in Each Quadrant," 19 June 1963.

P.E. Norsell, "Normally-Open Squib Controlled Valve in Bipropellant System," 19 June 1963.

P.M. Winslow, "Review of Marquardt Process Specifications--Syncom II," IDC, 15 June 1963.

J.C. Meyer, "Semiconductors Used on Syncom II Spacecraft," IDC, 19 June 1963.

L. W. Geary, "Component Failure Rate Improvements," IDC, 12 June 1963.

A. T. Owens, "Expected Value of the Period During Which a Noise Signal Exceeds a Given Threshold," IDC, 13 June 1963.

K.B. Lambert, "Syncom II Revisions of Preferred Parts, Materials, Processes Lists," 14 June 1963.

R.J. Smith, "Syncom II Power Amplifier Regulators," IDC, 14 June 1963.

R.A. Houghten, "Syncom II Coordination Control Summary Report," 14 June 1963.

P.S. Toorvald, "Syncom II Command Decoder Regulators," IDC, 13 June 1963.

J.G. Lotta, "Syncom II Mass Property Data," IDC, 10 June 1963.

A. T. Owens, "Report on GSFC Meeting Concerning Ground Systems Tests," IDC, 12 June 1963.

P. E. Norsell and J. W. Withers, "Syncom II Project Mass Properties Engineering Program," IDC, 11 June 1963.

R. Lindgren, "Syncom II Wiring Harness Design," IDC, 12 June 1963.

J.P. Wrzesinski, "Advanced Syncom Power Distribution," 7 June 1963.

R.A. Houghten, "Syncom II Coordination Control Summary," 7 June 1963.

"Syncom II Conference Report" (conducted at LMSC, Sunnyvale), 20 May 1963.

R.N. Ely, "Syncom II Solar Cell Test Program," IDC, 10 June 1963.

R.A. Houghten, "Syncom II Coordination Control Summary Report," 24 May 1963.

P.A. Rubin, "Minutes of GSE Meeting and Design Objectives for Advanced Syncom Command Generator and Telemetry Display," 5 June 1963.

J.M. Ferry, "Tank Connectors for Syncom II Bipropellant System," IDC, 5 June 1963.

J.P. Wrzesinski, "Syncom II Power Distribution," 4 June 1963.

A.J. McCall, "Recommended Tests of Syncom II Solenoid to Determine Electrical Interface," IDC, 31 May 1963.

"Mass Properties Control Program," Syncom II Project Notice No. 15, 24 June 1963.

"Statements of Work and Procurement Specifications for Syncom II Subcontracts," Syncom II Project Notice No. 14, 14 June 1963.